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Space Power

*Proceedings of a workshop
held at NASA Lewis Research Center
Cleveland, Ohio
April 10-12, 1984*



National Aeronautics
and Space Administration

Scientific and Technical
Information Branch

1984

PREFACE

Long range planning of research and technology programs in space power is essential to ensure that the appropriate technology will be available when needed. Vital to this planning process are the inputs of experts in the field, especially the collective view of such experts. The workshop process has been found to be a very effective way to elicit and crystallize these views.

In 1978 a highly successful workshop meeting was held at the Lewis Research Center on Future Orbital Power Systems Technology Requirements. This 1978 meeting was instrumental in focussing the NASA space power technology program to provide the technology base for the Space Station power system. When the initiation of the Space Station program appeared certain, it became timely in 1984 to hold another such meeting and look beyond the initial Space Station.

The Space Power Workshop was held in Cleveland at the Lewis Research Center on April 10-12, 1984. The objective of the Workshop was to explore appropriate directions for the applied research and technology programs that will enable the space power systems for the nation's future space missions beyond 1995.

The Workshop was arranged with the primary purpose of providing a forum for discussion among authorities on space power technology. Formal talks on missions, programs and state of the art were not primarily intended to disseminate new information but rather to refresh the participants' memories and to stimulate later discussions.

The Workshop explored the needs for advanced power systems technology for potential space missions within the public, military and commercial sectors. Plenary sessions were devoted to providing broad overviews of planned and potential missions within these three sectors and of present government research efforts in space power. These overviews were followed by presentations on the current status and trends in the various technologies that are encompassed by space power systems.

These plenary sessions set the stage for the "main event," the working group sessions. The participants divided into working groups that covered the principal disciplines involved in space power technology. Each working group was led by two cochairmen in informal discussions leading to recommendations for the appropriate directions for technology programs in their areas. In so doing, they examined the future needs, the present technology deficiencies, the opportunities in emerging technologies, and the adequacy and relevance of present technology programs. By all accounts, the working group sessions were lively and satisfying with a free and open exchange of opinions and ideas. A summary of the recommendations from each working group was presented and discussed in the final plenary session. After the meeting the cochairmen prepared final reports for each working group.

These Proceedings include the final working group reports together with the prepared talks that preceded the discussions.

Daniel T. Bernatowicz
William A. Brainard
Workshop Cochairmen
NASA Lewis Research Center

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PROPHETS FOR PROFITS FOR PEOPLE AND BUSINESS

Keynote Address

J. Stuart Fordyce
NASA Lewis Research Center

Let me add my welcome. We are gratified with the support from the government, commercial, and military sectors to this Workshop. The attendees, presenters, and working group chairpersons comprise the nation's foremost experts in space power technology. Over the next few days we will draw upon that expertise to help shape the NASA space power technology program which will greatly benefit from your individual and collective contributions. We hope you and your organizations will also carry away useful insights and ideas.

We are being asked to look to the future. As technologists we are motivated by the future moving just beyond our grasp. Figure 1 illustrates the vision of NASA's leaders. Dr. Payne was viewing the future from the Apollo perspective. Mr. Beggs' vision is at the threshold of what may be termed the third step, now that the Shuttle capability is in hand, the permanent manned Space Station. It was in 1978 that we met here to be prophets of just that future need. The workshop held then clearly concluded that high capacity space energy systems were critical. Technology programs were initiated and focused so that in 1984 we have available several power system options for consideration in the Space Station program and advanced development efforts are moving forward! In addition, there seems to be wide acceptance that power is a recognized enabling technology for the full exploitation of space. Also, we are seeing the move back into nuclear power for space taking place. It is, therefore, clear that the time is right to examine long range needs to try to solidify the technology thrusts needed to meet the next set of evolutionary objectives. This time, though, there is a qualitative difference which I attempted to bring out in the title of this talk. This can be seen clearly in the NASA goals (Figure 2) pertinent to this Workshop. The second goal develops a new theme; enhanced private investment and involvement in space. Aside from the overall federal policy environment, this arises from the recognition that NASA's financial participation in space is changing to a fraction of the total activity. I have used several charts that Dr. Jack Kerrebrock developed to articulate this point. The space market is growing rapidly (Figure 3). Several observations are made in Figure 4; foreign competition is formidable. In addition, NASA's charter includes responsibility to provide research and technology (R&T) support for military needs (Figure 5). One concludes that NASA's space R&T is in transition due to a change from a situation in the past where the NASA space program investment was commensurate with that occurring in the user community to a situation in which the investment will be a small fraction of the total (Figure 6). In the latter situation, NASA's space R&T effort will need to try to relate to the space "industry" in a manner similar to the way NASA's aeronautics R&T relates to the aeronautics industry.

This is articulated by the NASA Office of Aeronautics and Space Technology in a set of goals and objectives (Figure 7 and 8). To put flesh on these goals and objectives we must get your inputs on the challenges and exciting times ahead.

We must, in fact, match mission requirements to power system capabilities (Figure 9). Missions are ill-defined and power system capabilities are evolving, so many options must be carried along recognizing that lead times are long.

In our deliberations we will need to keep in mind the criterion of return on investment. Limited R&T resources must be invested where the biggest benefit can be expected and, since it is to be our policy to work closely with broader space interests, real world constraints of time and dollars and the skepticism of corporate boards of directors and financial analysis and the Office of Management and Budget must be taken into account.

In closing, let me observe that many of us here have been in this business for 25 years or more. The younger generation must be given the chance to be involved in what will shape the next 25 years, listen to them! Let's benefit from a slightly altered TRW institutional promotion, "I have an idea." Remember how fragile they are (Figure 10). Thank you again for your participation. We all await with anticipation the outputs on Thursday. Good luck!

"FUTURISTS"

BEGGS - MARCH 1984

.... FUTURE COULD INCLUDE A MANNED SPACE STATION IN LUNAR ORBIT WITHIN ABOUT 20 YEARS AND A MANNED COLONY ON THE MOON'S SURFACE BY ABOUT 2030 AN INITIAL STATION ON MARS UNDER CONSTRUCTION BY 2030 AND A MANNED COLONY BY 2060.

PAYNE - JULY 1969 PERSPECTIVE AT THE "FIRST STEP"

.... BY 1984 ROUND TRIP ROCKET PLANE TO COMFORTABLY APPOINTED SPACE STATION, COST SEVERAL \$1000 RT, A LUNAR SURFACE BASE AND A LUNAR ORBITING SPACE STATION, A NUCLEAR ROCKET SHUTTLE-EARTH TO MOON FOR \$10,000 RT, THE FIRST MEN LANDED ON MARS.

.... APOLLO 11 DEMONSTRATES THAT MAN CAN OPEN NEW WORLDS WHERE EVENTUALLY EXCITING NEW EXTRATERRESTRIAL SOCIETIES WILL BE FOUNDED.

Figure 1.

NASA GOALS

- CONDUCT EFFECTIVE AND PRODUCTIVE SPACE APPLICATIONS AND TECHNOLOGY PROGRAMS WHICH CONTRIBUTE MATERIALLY TOWARD U.S. LEADERSHIP AND SECURITY
- EXPAND OPPORTUNITIES FOR U.S. PRIVATE SECTOR INVESTMENT AND INVOLVEMENT IN CIVIL SPACE AND SPACE-RELATED ACTIVITIES

Figure 2.

SPACE MARKET... INTO ITS GROWTH PHASE

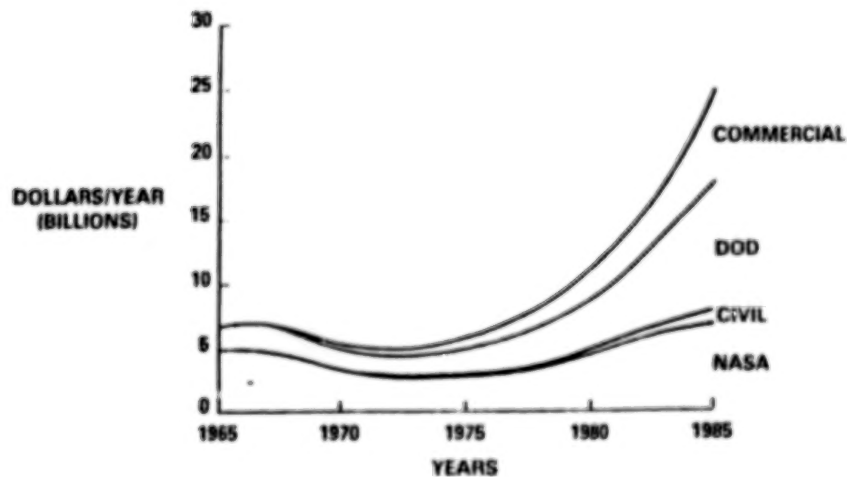


Figure 3.

SPACE MARKET OBSERVATIONS

THE MARKET IS CLEARLY INTO THE EXPONENTIAL GROWTH PHASE

- BASED ON KNOWN COMMUNICATIONS TRENDS
- OTHER EMERGING MARKET POTENTIALS

IN ADDITION TO THE DOMESTIC MARKET, ENORMOUS FOREIGN MARKET POTENTIAL

- MANY WITH NO COMPETING OPTIONS

INTENSIVE COMPETITION BETWEEN THE U.S. AND OTHER COUNTRIES HAS BEGUN

- LARGE FOREIGN R&D INVESTMENT
- STRONG COORDINATED NATIONAL EFFORTS

Figure 4.

MILITARY SUPPORT

- DOD EVOLVED TO CRITICAL DEPENDENCE ON SPACE ASSETS...
 - COMMUNICATIONS – INTELLIGENCE – NAVIGATION
 - RECONNAISSANCE – WEATHER
- MILITARY PLANNING INCORPORATES HIGHLY EXPANDED SPACE ACTIVITIES FOR...
 - SYSTEM READINESS, SURVIVABILITY, AND RECONSTITUTION
 - NEW AND IMPROVED CAPABILITIES
- NASA HAS A LEGISLATED R&T SUPPORT ROLE
 - AERONAUTICS ANALOGY

Figure 5.

SPACE R&T IN TRANSITION

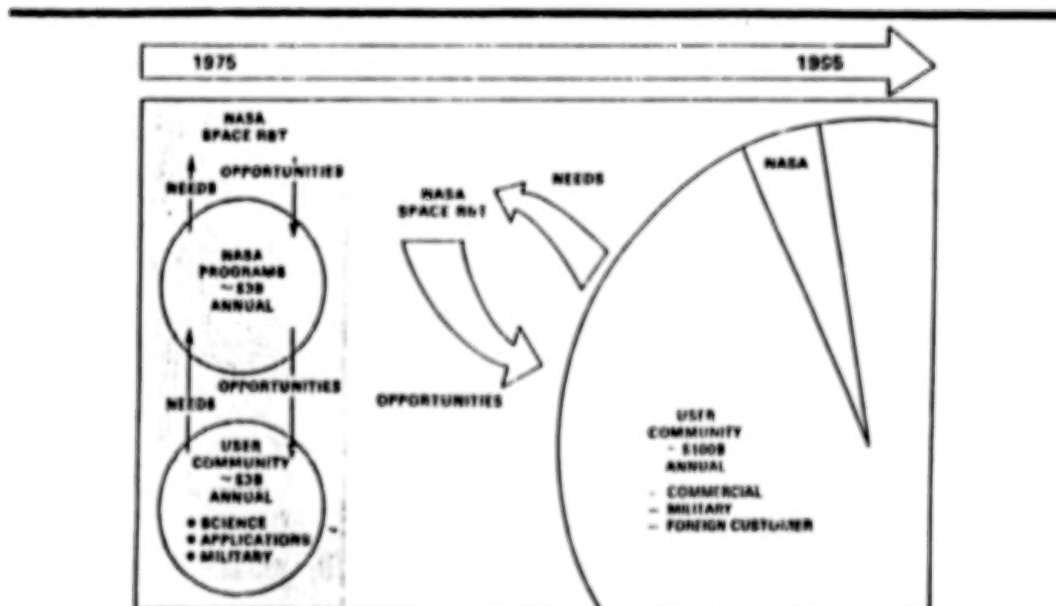


Figure 6.

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SPACE TECHNOLOGY LONG RANGE PLAN

GOAL

**STRENGTHEN THE AGENCY'S ADVANCED
SPACE R&T PROGRAM AS AN EFFECTIVE,
PRODUCTIVE, AND LONG-TERM CONTRIBUTOR
TO THE CONTINUED PREMINENCE OF THE
U.S. IN SPACE**

Figure 7.

SPACE TECHNOLOGY LONG RANGE PLAN

OBJECTIVES

- **PROVIDE U.S. R&T CAPABILITY BY MAINTAINING NASA CENTERS IN POSITIONS OF UNDISPUTED EXCELLENCE IN CRITICAL SPACE TECHNOLOGIES**
- **STRENGTHEN NASA'S SPACE R&T PROGRAM TO INSURE THE TIMELY PROVISION OF NEW CONCEPTS AND ADVANCED TECHNOLOGIES FOR THE U.S. CIVIL AND MILITARY SPACE ACTIVITIES**
- **ASSURE BALANCED PARTICIPATION IN SPACE R&T PROGRAM BY NASA CENTERS, GOVERNMENT AGENCIES, UNIVERSITIES, AND INDUSTRIAL RESEARCH ORGANIZATIONS**

Figure 8.

POWER SYSTEM AND MISSION MATCHING

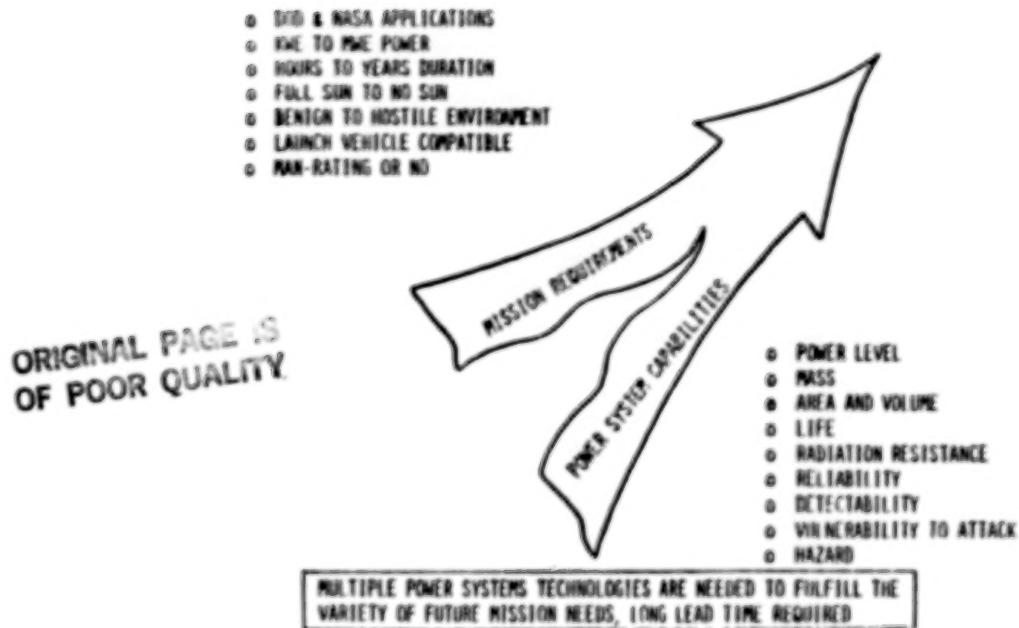


Figure 9.

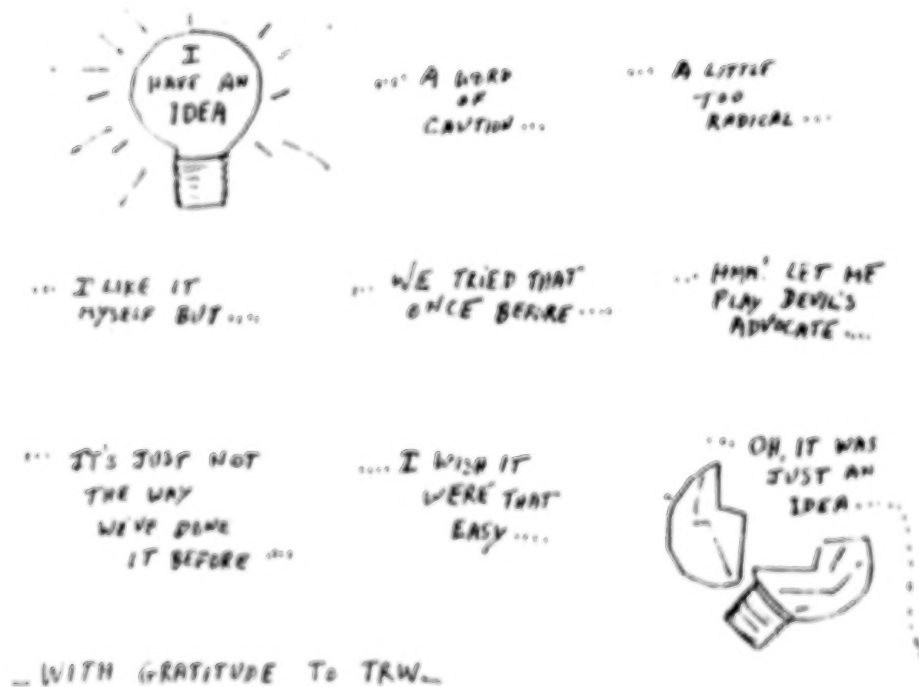


Figure 10.

SPACE STATION POWER SYSTEM

C. R. Baraona
Space Station Program Office
NASA Headquarters

This paper outlines the strategies, reasoning, and planning guidelines used in the development of the United States Space Station Program. The power required to support Space Station missions and housekeeping loads is a key driver in overall Space Station design. Conversely, Space Station requirements drive the power technology. Various power system technology options are discussed. The mission analysis studies resulting in the required Space Station capabilities are also discussed. An example of Space Station functions and a concept to provide them is presented. The weight, area, payload and altitude requirements on draft and mass requirements are described in this paper with a summary and status of key power systems technology requirements and issues.

Many Space Station power system technology options are available. However, the requirements for high power level, an initial operational capability in the early 1900's and the programmatic tendency toward low-risk approaches will strongly influence the options selected. Power system commonality within the diverse elements of the Space Station cluster including the manned base, unmanned platforms, orbital vehicles, and free-flying satellites will be difficult. The functional diversity of these elements influencing the power system technology away from commonality is counteracted by the programmatic need to conserve funds through commonality. Power system weight and drag make-up fuel weight are also prime considerations.

At this time, the status of the power system technology options is as follows:

- (1) Planar silicon arrays offer low technology risk but high weight and drag area.
- (2) Concentrator arrays promise lower cost and low drag area but increase technology risk.
- (3) Solar arrays that are erectable, deployable, or some combination of both are feasible approaches.
- (4) Earth-based, beamed energy transmission concepts studied to date are not viable options.
- (5) A solar dynamic system is a low drag candidate for the growth station and a high risk option for the initial station.
- (6) A nuclear system is a candidate for a growth station if safety, cost, and station configuration problems can be solved.

- (7) Nickel cadmium batteries have lower technology risk but high weight and high operational and system complexity.
- (8) Regenerative fuel cells are a promising energy storage option and will be technology ready for the initial station.
- (9) Inertial flywheel energy storage is a candidate for the growth station that offers benefits if combined with the attitude control system.
- (10) High voltage distribution (i.e. greater than 28 volts) is required but the cost and operational impact on the station of A.C., D.C. or hybrid systems is not yet a discriminator.
- (11) Power system modularity and transparency to evolving technology are mandatory.

The Space Station system engineering and integration activity will deal with these forces of user accommodation, schedule demands, program risk and funding constraints, and select technologies during the upcoming system definition and design.

A SPACE STATION ARCHITECTURE: CLUSTER CONCEPT

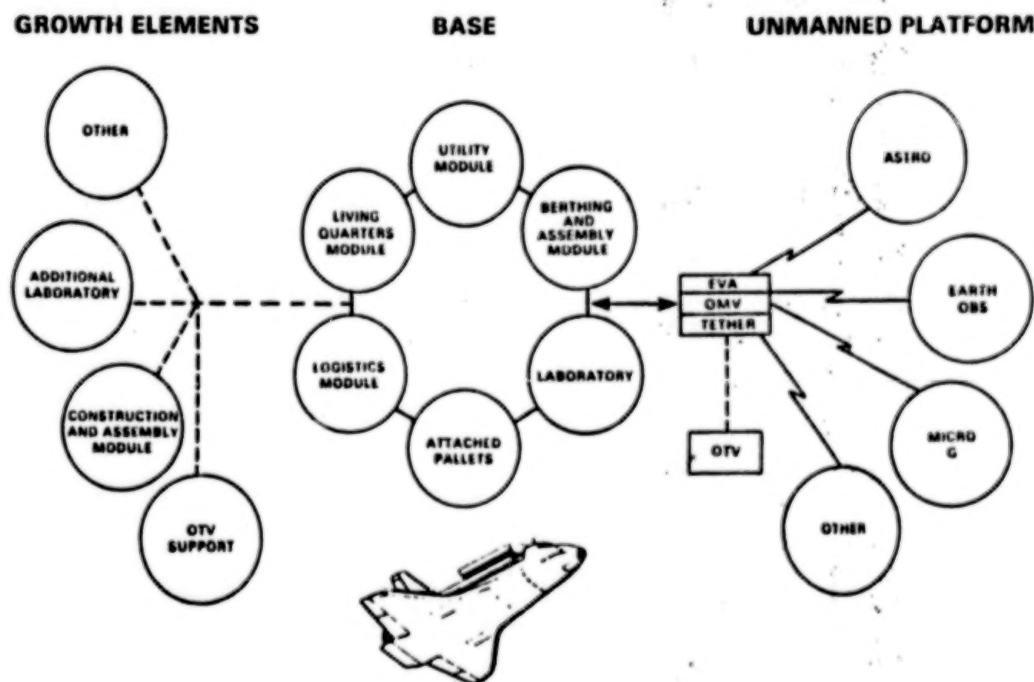


Figure 1.

FUNCTIONS OF A FUTURE SPACE STATION

- On-orbit laboratory
 - Science and applications
 - Technology and advanced development
- Permanent observatory(s)
- Transportation node
- Servicing facility
 - Free flyers
 - Platforms
- Communications and data processing node
- Manufacturing facility
- Assembly facility
- Storage depot

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A space station is a multi-purpose facility

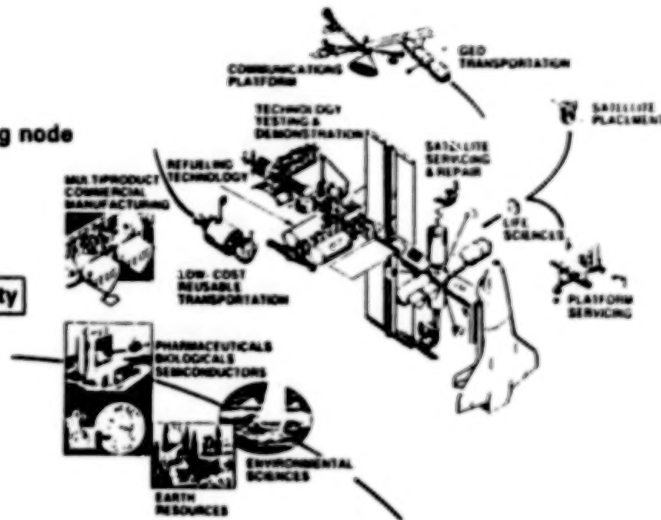


Figure 2.

SPACE STATION DEFINITION PRELIMINARY MISSION DATA BASE (1991-2000)

- Initial Data Base
- Derived from Shuttle and ELV Base
- Will Change as Station Capabilities Become Better Understood and Mission Priorities Shift
- Not the List of Mission/Payloads the Station Will Fly in 1991

SCIENCE AND APPLICATIONS

Astrophysics
Earth Science and Applications
Solar System Exploration
Life Sciences
Materials Science
Communications

COMMERCIAL

Materials Processing in Space
Earth and Oceans Observations
Communications

TECHNOLOGY DEVELOPMENT

Materials and Structures
Energy Conversion
Computer Science and Electronics
Propulsion
Controls and Human Factors
Space Station Systems/Operations
Fluid and Thermal Physics

Figure 3.

ESTIMATED SPACE STATION SYSTEM POWER REQUIREMENTS

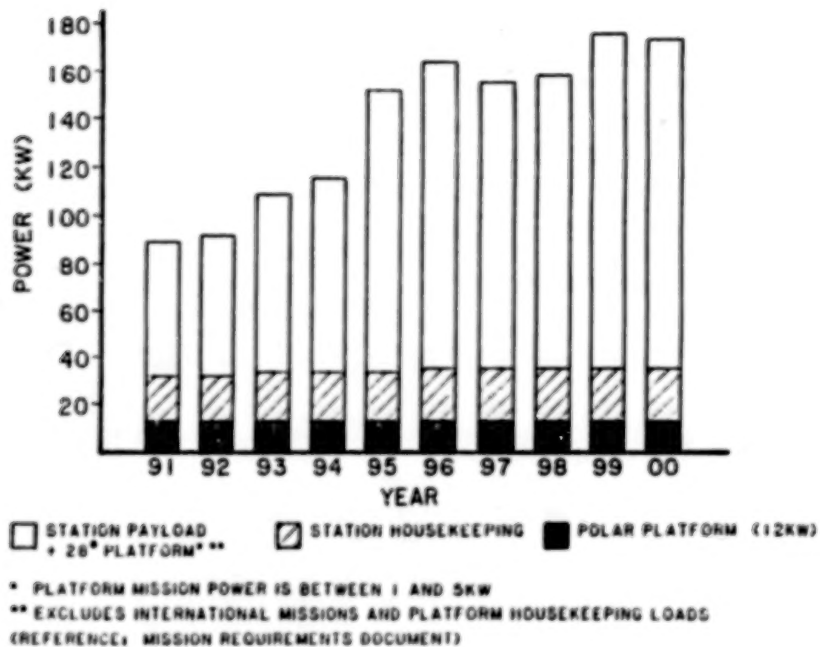
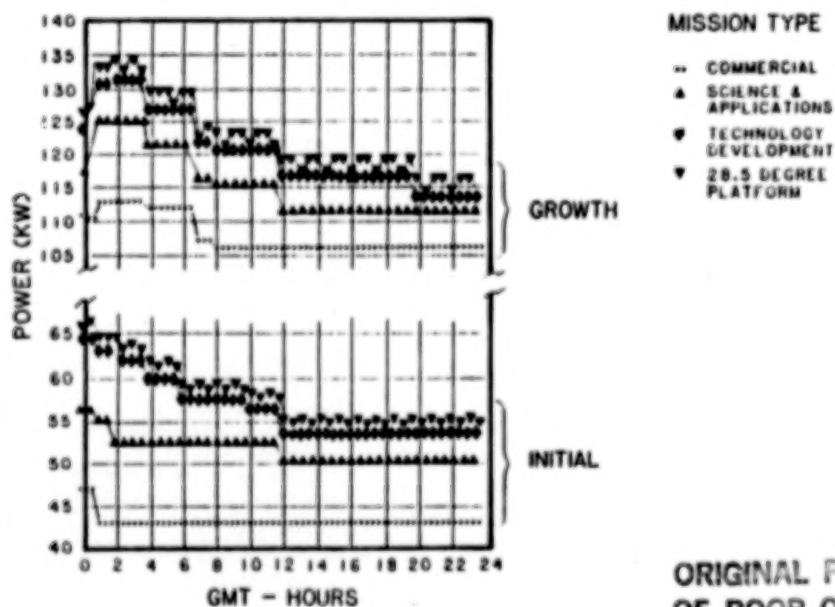


Figure 4.

DAILY PAYLOAD POWER PROFILES



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Figure 5.

REQUIREMENTS DRIVE POWER TECHNOLOGY

SPACE STATION REQUIREMENTS

- 75KW INITIAL
- 150KW GROWTH
- PERMANENCE OF SYSTEM
- EVOLUTIONARY GROWTH
- HUMAN PRESENCE
- OPERATIONAL FLEXIBILITY
- MULTIPLE USE
 - PAYLOADS
 - TRANSPORTATION CENTER/NODE
 - ORBITAL OPERATIONS

POWER TECHNOLOGY

- HIGH POWER LEVELS
- LARGE THERMAL LOADS
- LONG OPERATIONAL LIFE TIME, MAINTAINABILITY
- GROWABLE SUBSYSTEMS, MODULARITY
- SAFETY, CREW ACCOMMODATIONS
- GROUND INDEPENDENT, ON-BOARD MONITORING AND CONTROL
- USER FRIENDLY

Figure 6.

POWER SYSTEM DESIGN AND PERFORMANCE DRIVERS SIZE, COMMONALITY, GROWTH, MODULARITY

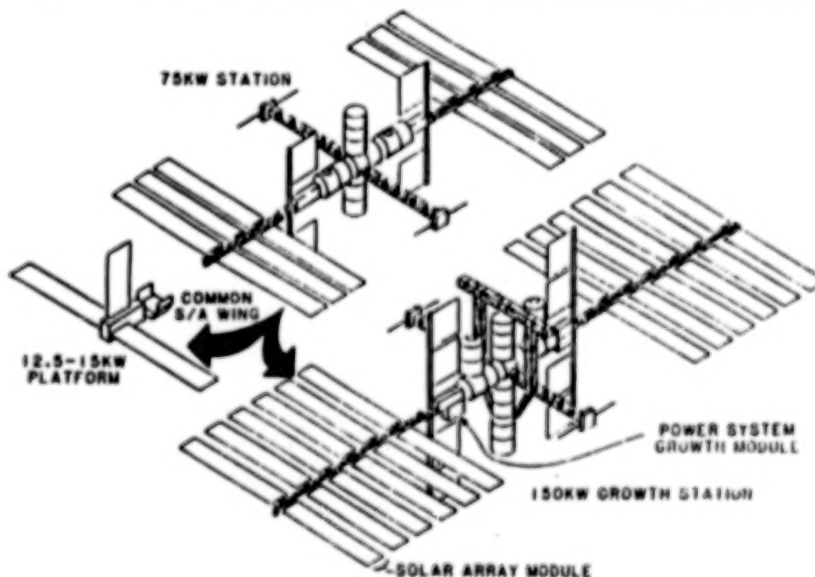


Figure 7.

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SPECIFIC WEIGHT AND AREA OF SEVERAL SPACE STATION POWER SYSTEMS

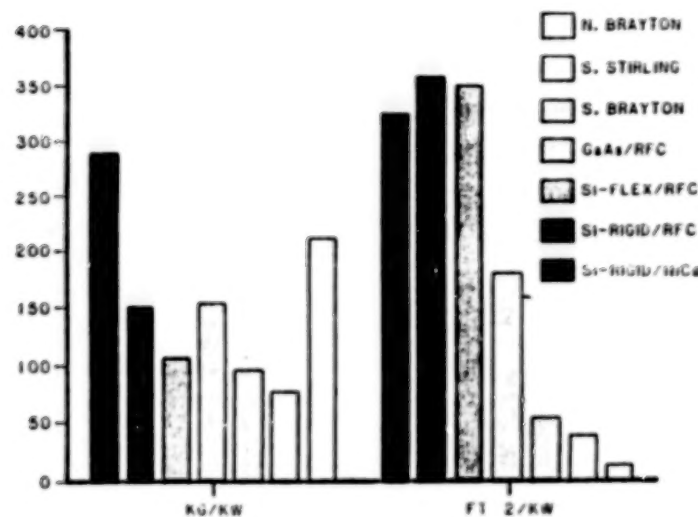


Figure 8.

10 YEAR ACCUMULATED PAYLOAD VS. ALTITUDE 75 KWe TO BUSS 28.5 DEGREE ORBIT +2 SIGMA DENSITY

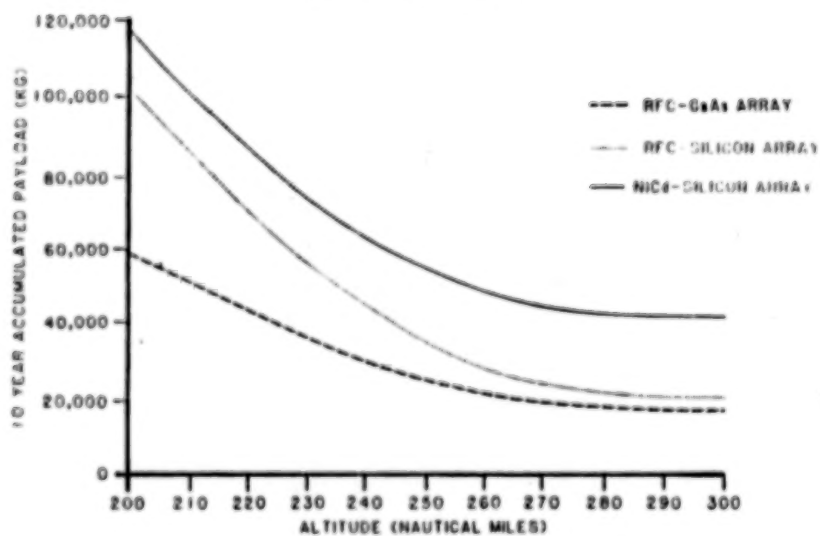


Figure 9.

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**MASS AND ARRAY AREA FOR 37.5 KWe SPACE STATION
SOLAR PHOTOVOLTAIC SYSTEMS WITH ERECTIBLE ARRAY**

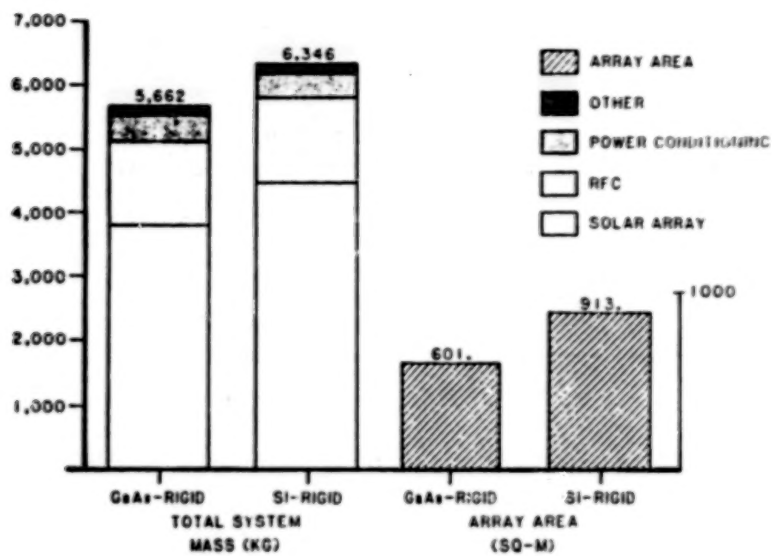


Figure 10.

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SPACE SCIENCE AND APPLICATIONS OVERVIEW OF PLANNED/POTENTIAL MISSIONS

W. L. Piotrowski
NASA Headquarters

No text was available at time of printing.

SPACE SCIENCE AND APPLICATIONS PROGRAM

- 0 STUDY OF DISTANT UNIVERSE
- 0 EXPLORATION OF NEAR UNIVERSE
- 0 EARTH AND ITS ENVIRONMENT
- 0 LIFE SCIENCES
- 0 APPLICATIONS
 - MATERIALS PROCESSING IN SPACE
 - COMMUNICATIONS

Figure 1.

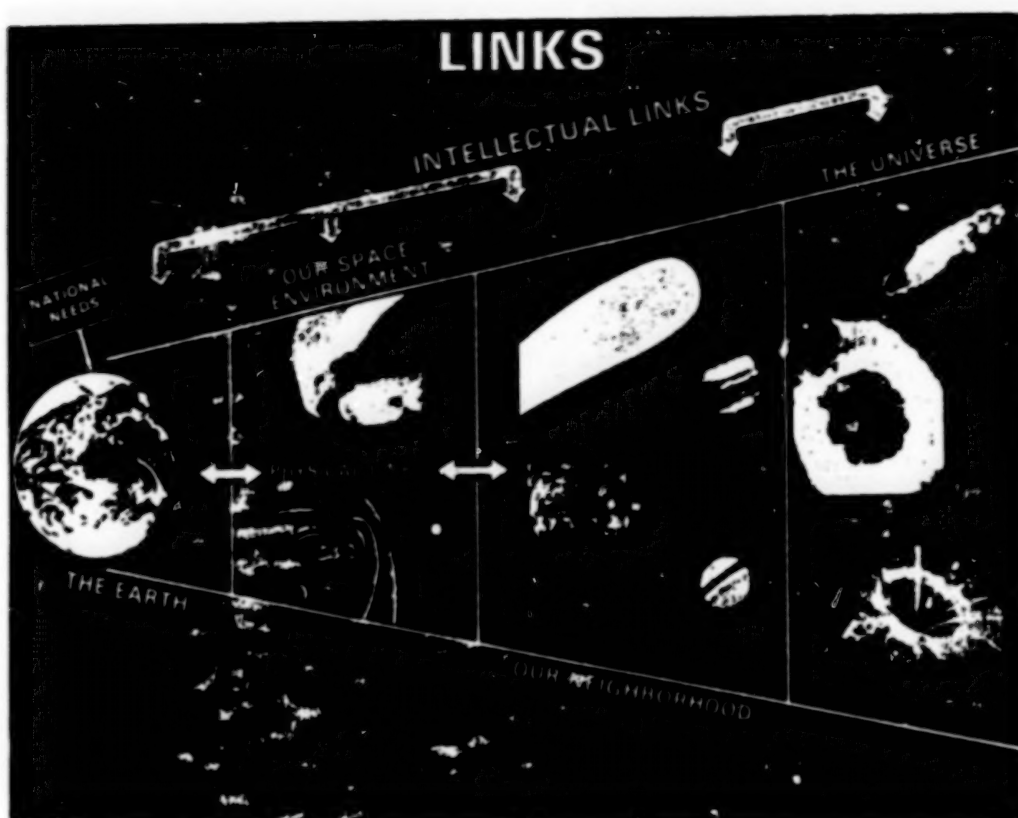


Figure 2.

SPACE SCIENCE AND APPLICATIONS

MAJOR FLIGHT PROGRAMS UNDER DEVELOPMENT

0 FAR UNIVERSE		
--	HUBBLE SPACE TELESCOPE	1986
--	GAMMA-RAY OBSERVATORY	1988
--	SOLAR OPTICAL TELESCOPE	1991
--	EXPLORERS (ROSAT, CORF, FIVE)	1986
--	SPACELAB 2	1985
0 NEAR UNIVERSE		
--	GALILEO	1986
--	INTERNATIONAL SOLAR POLAR MISSION	1986
--	VENUS RADAR MAPPER	1988
--	MARS GEOSCIENCE/CLIMATOLOGY OBSERVER	1990
0 EARTH AND ITS ENVIRONMENT		
--	UPPER ATMOSPHERIC RESEARCH SATELLITE	1989
--	EXPLORERS (AMPTE)	1984
0 LIFE SCIENCES		
--	SPACELAB 4	1986
0 APPLICATIONS		
--	SPACELAB 3 (MICROGRAVITY)	1984

Figure 3.

SPACE SCIENCE AND APPLICATIONS

PLANNING STRATEGY

- 0 BALANCED PROGRAM OF SCIENCE, EXPLOATION, AND APPLICATIONS
- 0 APPROPRIATE MIX OF EXPLORERS/OBSERVERS, MEDIUM-SCALE FREE-FLYERS, OBSERVATORIES, EARTH-ORBITAL PLATFORMS, AND SHUTTLE/SPACELAB MISSIONS
- 0 INCORPORATE SPACE STATION UTILIZATION IN PLANNING
- 0 FUTURE SPACE STATION/PLATFORM ENABLING MISSIONS
- 0 INTERNATIONAL COOPERATION/PARTICIPATION

Figure 4.

SPACE SCIENCE AND APPLICATIONS
POTENTIAL NEW MISSIONS

FY 1986-1987

- 0 OCEAN TOPOGRAPHY EXPERIMENT (TOPEX)
- 0 INTERNATIONAL SOLAR TERRESTRIAL PHYSICS (ISTP) PROGRAM
- 0 COMET RENDEZVOUS - ASTEROID FLYBY (CPAF)
- 0 SHUTTLE INFRARED TELESCOPE FACILITY (SIRTF)
- 0 ADVANCED X-RAY ASTROPHYSICS FACILITY (AXAF)

FY 1988-1990

- 0 GEOPOTENTIAL RESEARCH MISSION (GRM)
- 0 SYSTEM Z (FOS)
- 0 GRAVITY PROBE-B (GP-B)
- 0 SATURN ORBITER-TITAN PROBE (SOTP)
- 0 GEOSTATIONARY PLATFORM

(PLUS SPACELABS, EXPLORERS, PLANETARY OBSERVERS)

Figure 5.

OVERVIEW

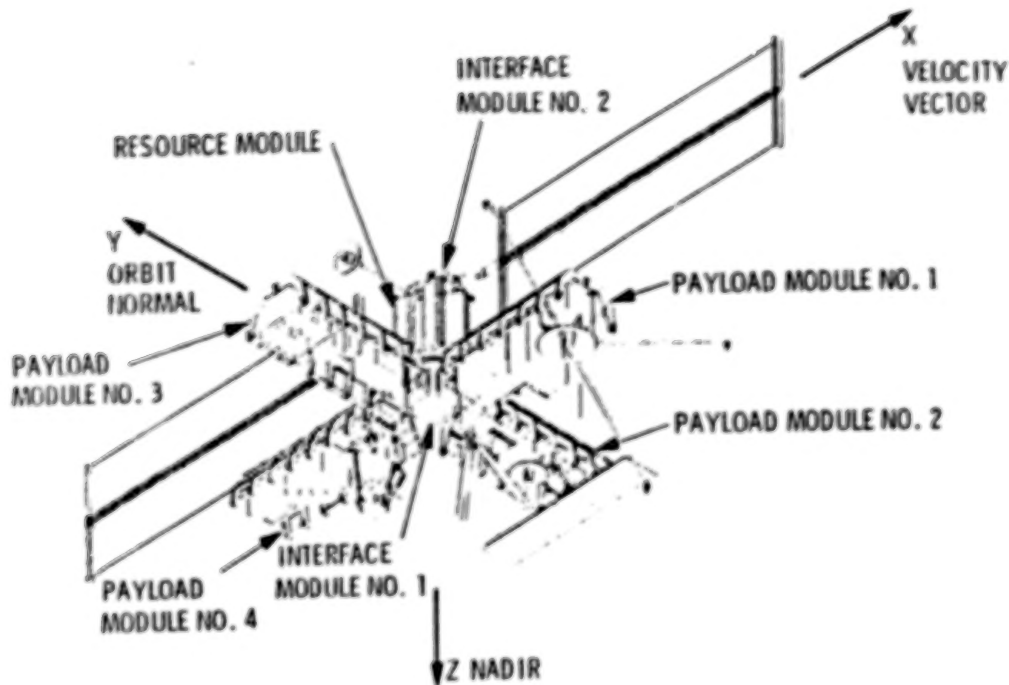


Figure 6.

SPACE SCIENCE AND APPLICATIONS

POST - 1990 MISSIONS

FAP UNIVERSE

LARGE DEPLOYABLE REFLECTOR
STAR PROBE
HIGH THROUGHPUT MISSION
ORBITING VERY LONG BASELINE INTERFEROMETRY

NEAR UNIVERSE

MAIN-BELT ASTEROID PENETRATION
JUPITER FLYBY/PROBE
NEPTUNE FLYBY/PROBE
MARS SAMPLE RETURN MISSION
COMET NUCLEUS SAMPLE RETURN

EARTH AND ITS ENVIRONMENT

WINDSAT
FIREX

Figure 7.

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Mars Sample Return Key Systems Technology Element

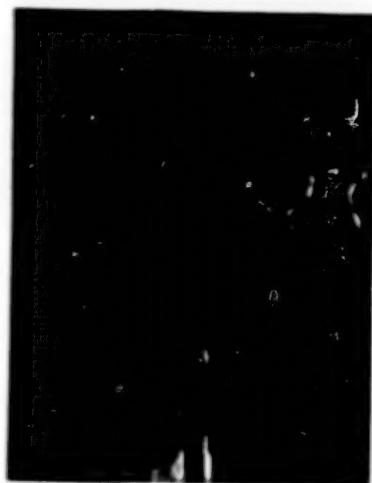
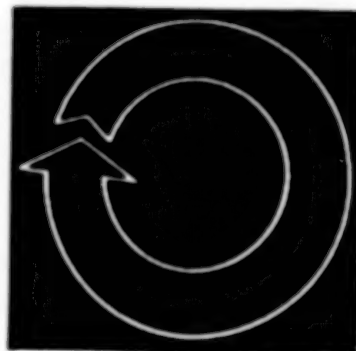


Figure 8.

Comet Nucleus Sample Return Key Systems Technology Elements

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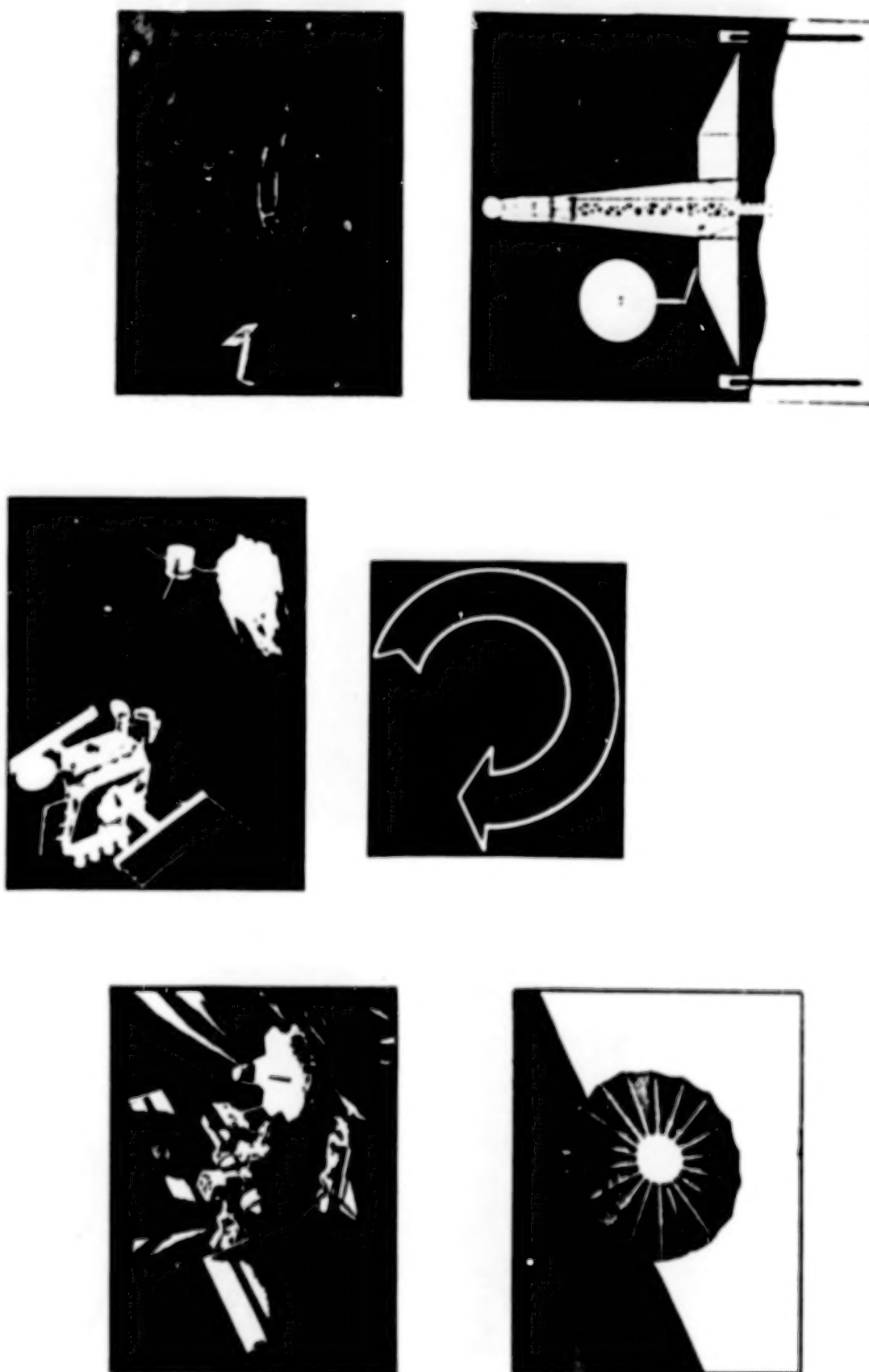


Figure 9.



Figure 11.

SPACE TO GOALS AND APPLICATIONS
COLLECTED FOR TERM MISSIONS

- 0 LEO OBSERVATIONS
 - FUNDAMENTAL PHYSICAL SYSTEM OF MOONS AND PLANETS CONCEPTS (ASTROPHYSICS)
 - ADVANCED SOLAR OBSERVATION
 - ASTROMETRIC SATELLITE
- 0 SPACE STATION LABORATORY
 - MICROGRAVITY LAB
 - LIFE SCIENCES LAB
 - PLANETARY PHYSICAL PROPERTIES LAB
- 0 ORBITATIONAL PLATFORMS
 - EARTH MONITORING SYSTEMS
 - COMMUNICATIONS
- 0 MANNED MOON MISSION
- 0 LUNAR BASE
 - RAD LINE OBSERVATION
 - CONTINUOUSLY MANNED STATION
- 0 SOLAR SYSTEM MISSIONS
 - OUTER PLANET ORBITERS/PROBES/LANDERS
 - INNER PLANET ORBITERS/LANDERS
 - PRIMITIVE BODY LANDERS

Figure 10.

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POWER TECHNOLOGY ISSUES

- 0 QUANTITY AND QUALITY
 - FROM WATTS TO KILOWATTS
 - SENSITIVE EXPERIMENTS NEED TENDER LOVING CARE
- 0 FLEXIBILITY
 - VARY THE SOURCES TRANSPARENTLY (TO THE USER)
 - PROVIDE VARIOUS CURRENTS AND VOLTAGES
- 0 RELIABILITY
 - OPERATE AUTOMATICALLY
 - EXPENSIVE MISSIONS CANNOT TOLERATE POWER FAILURE
 - FAULT TOLERANT
 - CLEVER REDUNDANCY
- 0 SEVERE ENVIRONMENTS
 - THERMAL EXTREMES
 - PRESSURE EXTREMES
 - CORROSION...WINDS, FLUIDS, SOLIDS
 - VIBRATION...ROCKETS

Figure 12.

TECHNOLOGY RESPONSES

- 0 QUANTITY AND QUALITY
 - WATTS/KG
 - WATTS/\$
 - WATTS/CM. INCH
 - INTERACTION CONTROL (INTERFERENCE)
- 0 FLEXIBILITY
 - CONFIGURATIONAL (MODULARITY)
 - USER INTERFACES
 - MANAGEMENT AND CONTROL (PROGRAMMABLE)
 - TECHNOLOGICAL ADVANCES
 - SOURCE INTERFACES
- 0 RELIABILITY
 - LIFETIME
 - AUTONOMY
- 0 SEVERE ENVIRONMENTS
 - MATERIALS
 - SHIELDING
 - PACKAGING

Figure 13.

POWER TECHNOLOGY ELEMENTS

- 0 SOURCES
 - STORED.....BATTERIES, FUEL-TO-ELECTRICAL?
 - NUCLEAR.....FUEL COSTS, CONTAMINATION
 - SOLAR.....SIZE, CONCENTRATION, MATERIALS
- 0 CONVERSION
 - EFFICIENCIES
 - RADIATION OF WASTE HEAT
 - MATERIALS
- 0 MANAGEMENT
 - CONTROL
 - DISTRIBUTION
 - HEALTH AND FAULT HANDLING
 - AUTONOMY

Figure 14.

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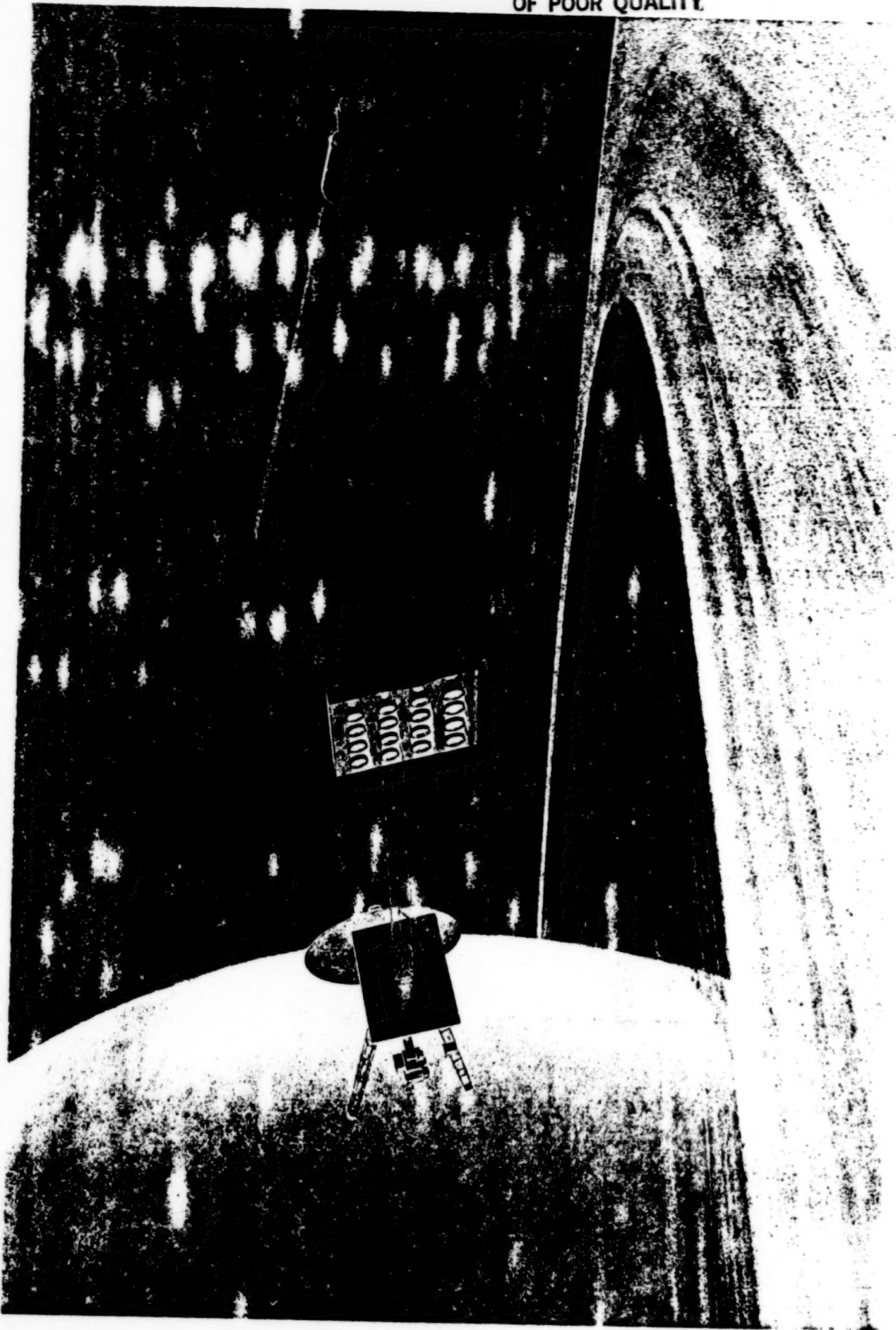


Figure 15.

INITIATIVES FOR THE POST-SPACE STATION ERA

Jesco Von Puttkamer
NASA Headquarters

The speaker discussed studies in process and shared recent thinking on initiatives for which a report is not yet available for distribution.

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EMERGING SPACE NUCLEAR POWER NEEDS

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Growing interest in new classes of military and civil space systems which demand substantial increases in power over current satellites has generated a renewed interest in space qualified nuclear power systems. Indeed, one can say that power is a limiting technology to the achievement of many of our future goals in space. Notwithstanding the general acknowledgement of this statement, however, the speed of nuclear power system development is currently limited by the lack of a clear distinct definition of system requirements.

BACKGROUND

There exists a rather broad misconception that space nuclear power is a new technology. In fact it is over twenty years old. Some twenty military space missions were flown utilizing space nuclear power systems between 1961-1977. Although these systems were generally of the Radioisotope Thermoelectric Generator (RTG) type, the launch of the SNAP 10A system in April of 1965 did place a 500 watt nuclear reactor system into space. Despite an unfortunate premature shut down due to a voltage regulator breakdown after only 43 days in orbit, its normal operation to that point plus the ground operation of an identical system for over 10,000 hours confirmed the technology. The emerging maturity and economics of photovoltaic/battery systems, however, prompted a general movement toward solar power systems as the backbone of earth orbital space programs while fuel cells were used for short duration manned missions. The requirement for long duration power independent of a solar source for interplanetary space explorations did continue to drive advances in nuclear isotope heat sources using thermoelectric power conversion.

REQUIREMENTS PULL/TECHNOLOGY PUSH

Although technologists are fond of planning technology limited programs, these programs most likely become funds limited somewhere early in the development cycle. The funding rate becomes dependent upon the total funds requirements versus funds availability and the acceptance of the system requirements defining the need for the new technology. While the promised gains from a new technology (technology push) can attract funds in the early, low cost stages of the development, eventually the ability to acquire sufficient funding to carry the program through the high cost advanced development stage depends upon widespread acceptance of a clearly defined set of system requirements. Competition for large amounts of funding is keen. Success demands convincing proof that accepted future missions cannot be implemented without the development and qualification of the needed technology.

The challenges facing the acquisition of sufficient funding for space nuclear power development are especially formidable. The sheer magnitude of funding projections places a particularly heavy burden of proof upon the technology program. Additionally, a naive but real tendency to equate nuclear power systems with nuclear weapons systems in space adds to that burden. Potentially the greatest challenge of all, however, concerns the safety of launching, orbiting and retrieving nuclear power systems. The strength of American public opinion regarding nuclear safety has convincingly demonstrated itself in the ground nuclear power business. Proponents of space nuclear power must be fully prepared to allay serious concerns for the consequences of launch failures, orbital decay and reentry, and possible retrieval and return of nuclear powered space systems.

The above described challenges heighten the difficulty of obtaining sufficient funds for the development of space nuclear power, but they do not make the task impossible. They do, however, place particular emphasis on the need for clearly articulated, well defined system requirements that can be uniquely satisfied by space nuclear power systems.

EMERGING SYSTEM REQUIREMENTS

From an intuitive viewpoint, the need for future space nuclear power systems is clear. It is generally acknowledged that even the most advanced solar power systems reach a practical limit between 30-50 KWe. System concepts with requirements exceeding this range are abundant. Requirements exceeding 100 KWe can only be satisfied with chemically driven or nuclear systems; power duration becoming the principle discriminator. The need for studies examining the tradeoffs between these two approaches to high power generation is apparent; however, an intuitive argument points out, for example, that even if the need for extremely high power is short (e.g., pulsed) the space system must operate for long periods in orbit during which some chemical constituents (e.g., cryogenics) will deplete due to boiloff.

Unfortunately, intuitive arguments don't often produce large funding. As pointed out earlier, the combination of large funding requirements plus the potential opposition due to perceived safety uncertainties places a large burden for clearly defined, accepted requirements upon the space nuclear power community. Potential systems that will produce such requirements do exist. They include the following:

- 1) Robust surveillance systems, both active and passive. While the survivability and performance of passive systems can be substantially enhanced with higher power, perhaps the most promising application is for space based radar systems. The specific requirements range from 50 KWe into hundreds of KWe depending upon specific mission applications and design configurations.

- 2) Survivable communications systems including anti-jam capabilities. Increased power enables more channels, higher frequencies, communication cross linking and anti-jam capability. Included within this group are blue-green laser communications systems. Potential power requirements approach 100 KWe.

- 3) Electric propulsion systems for reusable orbital transfer. High specific impulse, low thrust, reusable electric propulsion systems could be optimized around a nominal 100 day transfer from low earth to geosynchronous orbit. Payloads ranging to 30,000 lb could be transferred with a 500 KWe, 22 watt/lb nuclear power supply.

4) Weapons applications. Recent months have seen a rather vigorous effort within the DOD to define a development program for the President's initiative in Strategic Defense. Although virtually all of the proposed space based concepts require significant increases in power, in most cases specific requirements are closely tied to specific system configurations. Although broad statements (continuous power ranging from 10 to 150 KWe with semi-continuous to pulsed requirements in the multimewatt range) are available, they serve more to reinforce intuitive arguments than to determine specific requirements.

OBSERVATIONS AND CONCLUSIONS

Examination of the above classes of requirement leaves one with the feeling that they are just that - classes of requirements. Those who are beginning to define specific systems configurations in communications and surveillance are reluctant to define space nuclear power as an enabling technology because they are unsure of its availability. Weapons configurations are too early in the concept definition stage to clearly define specific requirements. Perhaps a more intriguing possibility emerges from an examination of a battle station concept which combines elements of all the above classes: weapons, surveillance, communication and propulsion. Such a concept could require a multi-modal high power source which could combine long duration low power operation for housekeeping functions with the capacity for multi-megawatt surges. Such a power source could also provide thermal energy for high thrust (nuclear rocket), rapid orbital transfer or electrical power for lower thrust, high specific impulse propulsion.

To date, specific systems requirements needed to attract the funding necessary for the immediate initiation of a large space nuclear power technology development program just aren't there. Accumulating evidence suggests, however, that such requirements are just over the horizon. This evidence lends credence to the intuitive conclusion that space nuclear power is an important key to a vigorous, robust future in space. Meanwhile, a vigorous technology effort must continue at a pace allowed by the appropriated funding. As Admiral Rickover is reputed to have said, "If the Navy had waited for the requirements process before building nuclear submarines, our submarines would still be diesel powered."

SPACE POWER FOR COMMUNICATION SATELLITES BEYOND 1995

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The NASA Lewis Research Center (LeRC) Space Power Technology Division has planned a Space Power Workshop. The objective of the workshop is to explore appropriate directions for the applied research and technology programs that will enable space power systems to meet the mission requirements of 1995 and beyond. In meeting the objectives of the workshop it is hoped that space power technology needs for potential space missions within the public, military and commercial sectors might be defined.

This paper has reviewed the space power trends for communication satellites beginning in the mid-70's. Predictions of technology advancements and requirements were compared with actual growth patterns. The conclusions derived from this trend study suggest that the spacecraft power system technology base and present rate of advancement will not be able to meet the power demands of the early to mid-90's. It is recommended that an emphasis on accelerating the technology development be made to minimize the technology gap.

BACKGROUND

In the early 1970's a study was performed for NASA, Ames Research Center to define key technology requirements for meeting the forecasted demands for commercial satellite services in the 1985 to 1995 time frame. This study was documented in 1973 in NASA document CR-114680, Technology Requirements for Post - 1985 Communications Satellites. The basic conclusions drawn in that study suggested that available (1972) and developing space power related technologies were adequate for filling forecasted 1985 to 1995 demands for service. Additionally, the 1980 to 1985 baseline spacecraft designs would probably fill the 1985 to 1995 time frame with minimum expected technology growth. This conclusion was based on using the same Delta Booster launch vehicle. Conversion to STS launch vehicles would result in approximately a 50% weight growth to allow for reduced development and manufacturing costs. It was felt that there existed a reduced need for technology improvements in structures, power and attitude control when converting from Delta to STS launch vehicles.

The technology requirements study recommended emphasis be made in the area of antenna technology and solid state transponder technology. A sustaining interest of about \$10 million per year was recommended in the power systems and other spacecraft system technology areas. These recommendations suggest that with a primary focus on antenna and power amplifier technology development the supporting systems like power, attitude control, propulsion and telemetry and command will naturally provide an adequate technology base for an advanced communications satellite system in the time period 1985 to 1995.

POWER SYSTEMS REQUIREMENTS

The study developed a list of 32 different needs and grouped them into seven categories for the 1985 to 1995 time period. These categories along with some of the 32 needs that currently have high visibility are listed below.

1. Television Service
 - Teleconferencing
 - Public Broadcast
 - State Government
 - Home TV (Commercial Broadcast)
2. Digital and Voice Service
 - Public Telephone
 - Business
 - General Computer Services
 - National Law Enforcement
 - Emergency and Disaster
 - Electronic Mail
 - Electronic Publishing
3. High Speed Data Relay
 - High Speed Computer Network
4. Mobile Services
 - Marine
 - Aircraft
 - Ground Vehicle
5. NASA Tracking and Data Relay Network
 - NASA Space Operations Network
6. Earth Resources Data Collection
 - Earth Resources Data Relay
7. RF Environment Monitoring
 - RF Environment Monitoring

The categories which lend themselves to commercialization, namely 1 through 4, have needs which are currently being satisfied or will be before the end of the 1980 decade.

Communication subsystem technology advances have been rapid in the 1975 to 1985 period. As predicted, the first solid state power amplifier was built (by RCA) in 1982 and the first all solid state communications subsystem was built by RCA and launched in 1984. The concept of direct broadcast has also achieved early maturity with service beginning in the middle of the 1980 decade.

Commercialization of space based communications is established and is accelerating. A good measure of this growth is the demand for power to operate the communications payload as seen in Figure 1. In the 1975 to 1985 period power requirements were steady with little growth. However, in the mid-1980's the power requirements started to accelerate and within 10 years (1995) the required power is expected to

be an order of magnitude higher. This growth is attributed to the emphasis placed on accelerating the communications technology development activity.

The power demands forecasted push the power subsystem technology base to its virtual limits. Since the demands are real up to the early 1990's the technology base and growth rate is adequate. However, beyond 1990 the power demands basically exceed nominal power system growth expectations.

A first step was taken in 1983 to change the basic power subsystem operational philosophy to meet the high power demands. A dual voltage unregulated bus system was designed for an RCA built Direct Broadcast Satellite (DBS). The spacecraft housekeeping requirements remained on a low voltage bus (28 to 35Vdc) while the separate payload bus was designed for 100Vdc. The high voltage is well within the current solar array state-of-art design technology. The resultant effect allowed for reduced power losses when converting the 100 volt input voltage to a required 7500 volt for TWTA operation and, therefore, a smaller solar array area. As the power demands increase so will the voltage requirements as seen in Figure 2. An upper limit of 300 volts dc is expected due to current power transfer and solid state device technology limits.

The solar array area will grow to meet the payload power demands. This growth is seen in Figure 3 for a deployable rigid one degree of freedom solar array. By the early 90's the STS one-quarter cargo bay limit will be met. This point in time can be extended out a few years by using more efficient solar cells. However, current silicon solar cell conversion efficiency values are nearing a practical limit of 14 to 15%. The GaAs solar cell is the next real candidate with efficiencies of 16 to 18%. This cell has not achieved a production status and projected costs are still high on a performance based comparison to silicon.

The predicted end-of-life (based on 10 year missions) solar array power requirements are seen in Figure 4. This data suggests that by 1995 the solar array technology base for communication satellites must be able to deliver from 6K to 11K watts of power for use during the following 10 year period. Since the solar array could degrade based on present day technologies up to 25% by the 10th year a beginning of life power capability of 7.5K to 13.8K watts is required. These power levels using conventional solar cell technology equate to tremendous area requirements. Technology bases for the solar array and other subsystem will need to be established to meet packaging articulation in orbit, power dissipation and transfer, and storage and deployment requirements.

Extreme power demands are also being forecasted for the battery system. As can be seen in Figure 5, the eclipse load requirements make an order of magnitude change from 1975 to 1995. The battery technology base is capable of meeting these demands into the late 80's with Nickel-Hydrogen (Ni-H₂) and high energy density Nickel-Cadmium. Both battery systems will be pushed to the absolute limits of depth-of-discharge and performance lifetimes.

Battery weight is a serious problem relative to a communications satellite. Since the battery is the heaviest power system component in terms of specific energy its use to support payload powers during the two yearly eclipse seasons impacts the ultimate weight of the communications payload and therefore the earnings ratio of the satellite system. High voltage requirements lead to increased reliability problems when large numbers of storage cells are placed in series. Large battery systems also create enormous thermal control problems due to the

distribution of the numerous battery packs on the spacecraft structure and the system requirements for minimal temperature differences between packs.

Power distribution and control bring the power system solar array and battery components together to support the payload and maintain the spacecraft housekeeping functions. Extreme high powers and higher operating voltages are real and serious challenges to the developing base. Major design changes and/or technology improvements will be required to meet the demands of the early 90's.

SUMMARY

As can be seen from the discussion and data trends presented, the communications satellite power system technology base is seriously challenged to meet the power demands of 1995 and beyond. In fact, there is a problem in satisfactorily meeting the power demands of the early 90's. This problem is primarily due to the advanced technology base established for the communications system in the early 80's and the subsequent rapid commercialization of that technology. The 1972 study on communication satellite technology requirements was on-target with technology assessments and trends. However, commercialization has literally pushed the clock ahead and provided an advanced communications technology base for both government (military and civilian) and private sections to use. As a result the normal development trends of spacecraft technologies will fall short of meeting the 1995 needs. Therefore, an accelerated AR&T effort in spacecraft technologies, especially power systems, must be undertaken. The predicted 1995 communications satellite power requirements can only be met if other spacecraft systems are ready in the 1990 to 1993 time period.

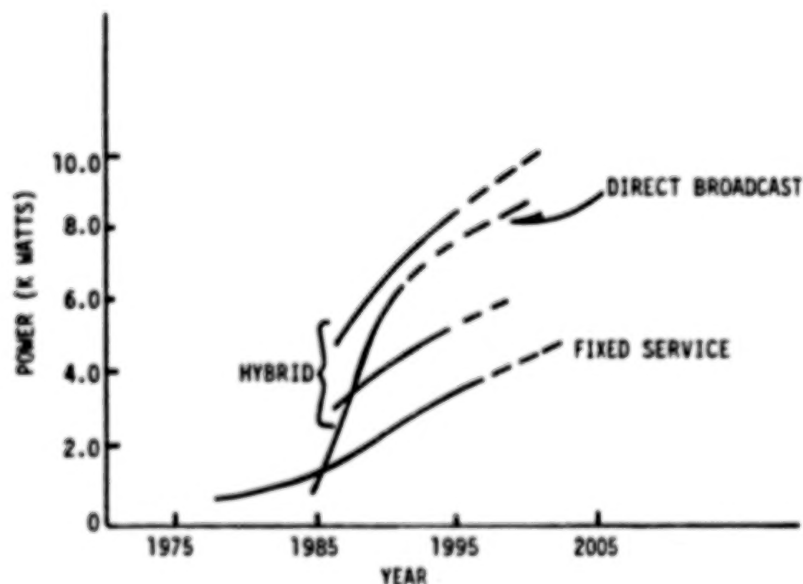


Figure 1. Communications Satellite Payload Input Power

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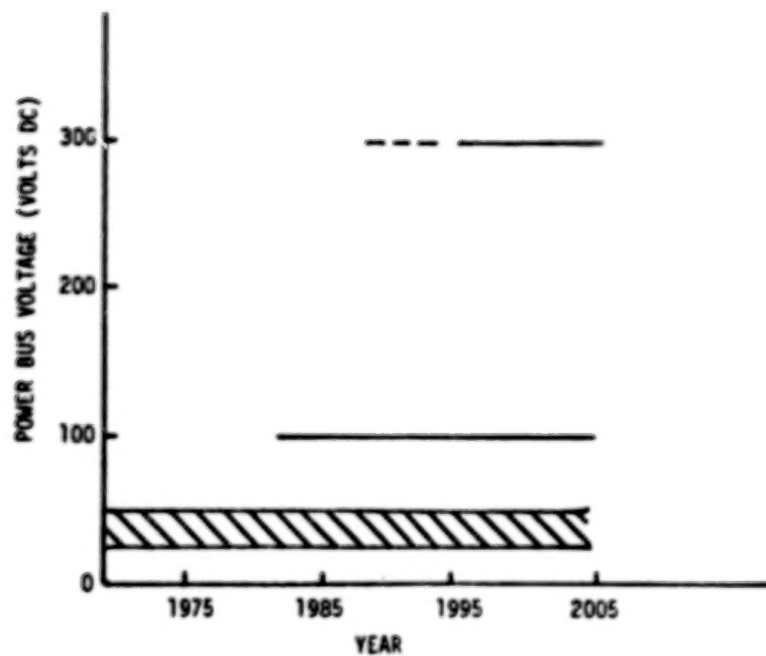


Figure 2. Voltage Trend Unregulated Bus

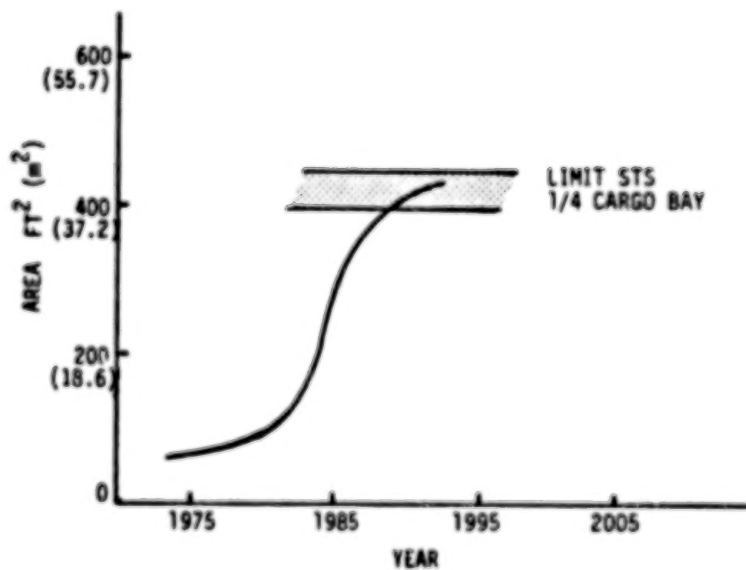


Figure 3. Communications Satellite Solar Array Area Growth, Flat, Rigid, One Degree Freedom

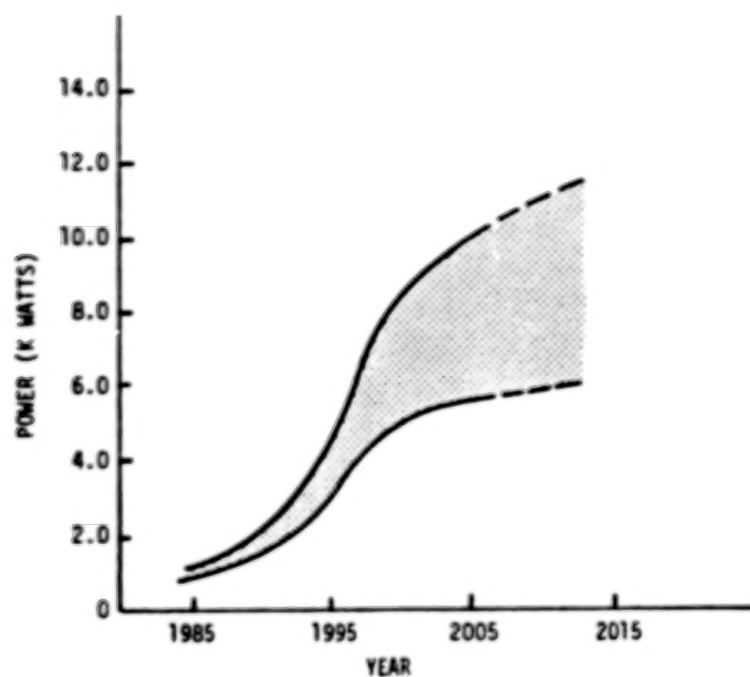


Figure 4. Communications Satellite Solar Array Power Requirements EOL (10 Yr)

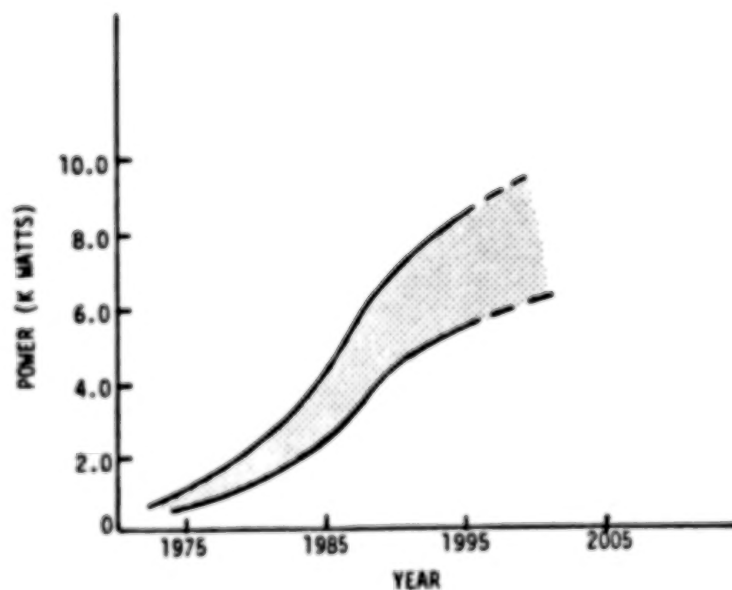


Figure 5. Communications Satellite Eclipse Operation Power Requirements

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POWER FOR COMMERCIAL SPACE APPLICATIONS

Philip K. Chapman
Arthur D. Little, Inc.

Since the proposed space station is intended to be a permanent installation, used in part by commercial organizations, its design requirements differ fundamentally from those in previous manned spacecraft. In particular, commercialization on a significant scale depends critically on the ability to control capital and operational costs, including the cost of energy, and this demands new approaches to systems such as the power supply for the space station. These considerations suggest guidelines for power plant development.

COMMERCIAL INTEREST IN SPACE

After many years of uncertainty and indecision, the principal elements of the U.S. space program for the next decade have recently become clear. The NASA permanently-manned space station (PMSS) and the development of ballistic missile defenses (BMD) will inaugurate a mature phase in the utilization of space, involving extensive operations and long-term habitation in orbit. If appropriate Federal policies are adopted, these developments may also permit the initiation of a broad, self-supporting commercial space program.

Over the past several years, NASA has made a commendable effort to obtain input from industry so that the PMSS may be designed to meet commercial needs. Many corporations have expressed a general interest in space applications, but few are as yet ready to consider significant investments in this new field. With some exceptions, corporate planners (in industries other than telecommunications) still see little profit potential in commercial space operations, so they have not considered in any depth what PMSS features might be of interest to their companies. While it may be hoped that this situation will improve with wider recognition of space station capabilities, new approaches are urgently needed if the resources of space are to be developed by free enterprise.

The technical, economic and institutional questions involved in space commercialization may be well illustrated by considering the problem of supplying electric power to commercial users of the space station. For definiteness, a photovoltaic power supply is assumed here, but similar considerations apply to other options such as nuclear power and electro-mechanical energy conversion.

COSTS AND COMMERCIALIZATION

A principal goal of Federal space policy must be to create conditions under which commercial space ventures can become viable investments by the customary standards of the private capital market. In particular, the costs and the downside risk must both be reduced to acceptable levels.

The policy of the present Administration is to permit commercial operators to use national space facilities at the marginal cost to the Government of such use. For example, NASA prices shuttle launches by treating STS development as a sunk cost, a national investment whose amortization need not be shared by commercial users. Under present planning, the PMSS will also be a facility which is developed and assembled at public expense. The prices charged to commercial users (e.g., to rent space aboard the station or to buy energy from the power supply) will not reflect amortization of the initial PMSS program cost, which is now estimated as \$8 billion.

This policy is certainly justified where space facilities are developed for national purposes other than commercialization (e.g., defense). It may also be justified where the explicit purpose of the facility is to develop commercial applications, as long as the implied subsidies serve the national interest and not particular companies. Federally-funded research facilities, serving broad segments of the U.S. economy, can provide benefits which are unavailable with other types of subsidy (e.g., tax incentives). For example, the PMSS may serve as a national laboratory, permitting the intriguing but uncertain prospects for new commercial products to be explored in ways which would not be possible for an individual corporation. Furthermore, the PMSS can and should be a demonstration project, encouraging investment by showing unequivocally that specific commercial space applications are feasible. In this connection, it must be remembered that the technical feasibility of a proposed application is of little interest to investors unless economic feasibility can also be shown.

Spaceflight remains an expensive venture, with launch costs to low Earth orbit (LEO) in excess of \$2400/kg, using the STS. In order to reduce transportation costs, the primary requirement is not a radical advance in booster technology, but a substantial increase in the freight throughput to orbit. When the traffic to orbit reaches thousands of tons per year, economies of scale will permit a reduction in launch costs by at least an order of magnitude (refs. 1, 2), using shuttle-derived vehicles or other heavy-lift designs. At present, there is little traffic to orbit because launch costs are too high; and costs are high because there is too little traffic to orbit.

Breaking this vicious circle is an important national objective, which requires synergism between civilian, military and commercial space projects. The Federal Government can aid the transition by supporting technical development, by increasing Federally-funded traffic to orbit (both military and civilian), and by taking steps to encourage commercial traffic.

For the next decade, commercial space products (other than information) will be limited to those few which are of sufficient value to justify the present high costs -- but the eventual contribution of entrepreneurs to the development of space applications depends on the achievement of much lower

costs and a much larger scale of operations. In the long run, space products must compete on equal terms with terrestrial alternatives. Even in those cases where the space product offers a truly unique benefit, the available market will depend on competing demands for the customer's dollar. Since the prices which can be charged for space products will be determined largely by exogenous market forces, there is for each product a definite limit to the acceptable manufacturing cost. Federal subsidies may be essential to fledgling space enterprises for the immediate future, but cannot be sustained indefinitely. The investment climate will be greatly improved if truly self-supporting commercial space operations are a realistic prospect.

In order to allow assessment of trends in space production costs, the accounting procedures used by NASA should be designed to provide a clear separation between R&D costs and the capital costs for construction. During the fabrication of PMSS hardware, the costs incurred should be analysed in order to show how economies can be achieved as the PMSS program matures towards full commercial involvement. Wherever possible, R&D efforts should be justified in terms of long-range objectives and funded independently of the PMSS project.

It should be noted that the cost of launching the initial PMSS to orbit (in four shuttle loads) is less than 4% of the total program cost. This suggests that cost-reduction efforts are well justified, even if they lead to a heavier station, more expensive to launch.

THE COST OF ENERGY IN SPACE

The current cost of a photovoltaic array (single-crystal silicon cells) for use in space (e.g., on a comsat) is of order \$500 per peak watt (\$500/Wp). Lower prices may be achieved as development costs are amortized, especially for larger arrays such as that for the PMSS. Gallium arsenide cells (2cm X 4 cm, 17% efficiency) are expected to be available soon at prices around \$100 each; GaAs arrays, with concentration, will then cost approximately \$500/Wp. For comparison, terrestrial photovoltaic arrays may be purchased, in quantity, at prices below \$10/Wp; the goal of the National Photovoltaic Program, administered by the Department of Energy, is to produce arrays at costs below \$1/Wp (ref. 3). Some of the reasons for the much higher price of space-rated arrays are discussed below.

Because of distribution inefficiencies as well as the day-night cycle in orbit, the peak output of the PMSS array must be about 2.7 times greater than its mean output. At \$500/Wp, the overall cost of the array would thus amount to \$1350 per average watt (\$1350/Wa). If the mass of the array is 5 kg/kWp, the launch cost (@ \$2400/kg) will be negligible by comparison, amounting to \$32/Wa. If a commercial organization were to purchase an array at this price, capital amortization* would contribute about \$90 per kilowatt-hour (\$90/kWh) to the cost of energy. This is at least three orders of magnitude more expensive than the cost of electric energy from conventional terrestrial

*Over 5 years at a discount rate of 13%; 50% load factor.

sources -- and it does not include the capital costs of the distribution system, energy storage for darkside power, etc., nor operational costs for maintenance, drag make-up, etc.

The conclusion from this simple analysis is that the true cost of energy in the PMSS, as presently configured, will be considerably in excess of \$100/kWh, perhaps over \$200/kWh. Even though the initial PMSS customers may pay discounted prices, full cost recovery will almost certainly be required before commercial operations become routine. It is therefore important to determine whether costs of order \$100/kWh may be a deterrent to these applications.

Table I lists the specific energies required for several representative physical processes, and the corresponding costs (@ \$100/kWh). For the most energy-intensive industrial processes which might be adapted for use in space, these energy costs can be very significant. In the last case listed (the Czochralski process for drawing a single-crystal silicon boule from a melt) the terrestrial process is very inefficient, and alternative processes may be preferred in orbit. Moreover, it may be possible to drive some processes with solar heat, rather than with electricity. Nevertheless, the examples given are sufficient to show that, in some processes, energy costs in the PMSS may exceed launch costs for the materials involved.

As noted above, launch costs will decrease as traffic to orbit grows. Energy costs must also decrease with time if they are not to become a serious impediment to the growth of commercial space enterprises. In order to demonstrate the eventual feasibility of large-scale space commercialization, it would be very useful to define a development path which can lead within a reasonable time to true electric energy costs, delivered in orbit, below \$10/kWh. Subsidies may be necessary in order to approach this price in the initial PMSS, but one of the important roles of the station will be to serve as a laboratory for the development of more economical power supplies.

GUIDELINES FOR POWER SUPPLY DEVELOPMENT

The target proposed here (\$10/kWh) involves a reduction in solar array and other capital costs by an order of magnitude, compared to current space-rated systems. The possibility of such a reduction depends on the fact that the space station differs greatly from previous space systems in both its structure and purpose.

There are many reasons for the present high costs of space hardware, including the following:

- High launch costs encourage miniaturized, high-performance equipment.
- For a specified mission, the chosen launch vehicle may impose strict limits on payload mass, and hence demand miniaturized equipment.

- Stressful environments (e.g., atmospheric entry) may require high performance and advanced technology.
- Development costs must be amortized over a small number of flight systems.
- Small production runs preclude economies through automated fabrication, learning effects, etc.
- Post-launch inaccessibility of spacecraft on long-duration missions demands high equipment reliability.
- Space-rating and especially man-rating to NASA standards involves very detailed and expensive testing and documentation.
- System failures may be unacceptable because of non-economic penalties (political factors, etc.).
- Success in NASA and DoD missions is judged by achievement of mission objectives, not by profit or loss.

It is not yet widely appreciated that none of these factors need apply to much of the equipment required for commercial purposes in the PMSS.

Although launch costs remain high, for many important PMSS systems (e.g., the solar array) they are negligible compared to present hardware costs.

The PMSS is not a spacecraft. Aerospace vehicles have always been designed for high performance in stressful environments. The PMSS is fundamentally different in that it is not intended to be maneuverable at all. It is much more appropriate to think of the station as a building (or perhaps a complex of buildings, as in an industrial park) than as a vehicle. Space station engineers are working in the construction industry, and the design philosophy which was so successful in Apollo and the STS is not applicable. The PMSS will last a long time, growing and changing gradually as new features are added and obsolete equipment is cannibalized or mothballed. Some form of configuration control will be needed (to maintain stability under gravity gradients, to avoid excessive structural loads during docking or other activities, and perhaps to control atmospheric drag), but basically there is no limit to the eventual mass of the system.

The solar array for the initial PMSS will have an output of 200 kwp. This is large by previous spacecraft standards, and should permit some economies of scale in production of the array. More importantly, space commercialization must grow if it is to succeed, which implies an increasing need for energy. A firm commitment to free enterprise in space (perhaps including some form of Federal guarantee for the required investment) may justify installation of a continuing production line for array modules.

Terrestrial photovoltaic array designs (especially those using single-crystal silicon cells) are often massive by space standards, and their performance may deteriorate rapidly in the presence of ionizing radiation. The space environment also poses other problems, such as the erosion of

kapton film and other materials by residual monatomic oxygen in LEO. For these reasons, terrestrial cells typically cannot be used without modification in LEO, especially if the orbit passes through the South Atlantic Anomaly. However, cells designed for use in space could be adapted to terrestrial use, if they were cheap enough. A cooperative program might permit significant cost reductions, if it were aimed at developing a class of photovoltaic cells which could be adapted with relatively minor changes to both space and terrestrial applications. Thin-film cells which are now under development may be good candidates for such a program, since they promise both low mass and good radiation resistance.

PMSS systems should be designed to commercial quality standards. The traditional NASA approaches to system reliability (multiple redundancy, zero defects, etc.) are appropriate only for safety-critical systems or those whose failure could endanger the station. The PMSS will undoubtedly need a highly reliable back-up power supply, with sufficient output to maintain life support and other critical functions in the face of the maximum credible emergency. However, the main power supply, used for industrial processes, could take advantage of reliability assurance procedures used in terrestrial power systems (e.g., system reconfiguration to bypass failed elements). Most commercial users would much prefer to pay \$10/kWh, with an outage every few months, than \$100/kWh for power which never fails.

In general, systems should be maintainable rather than reliable — but, because of the personnel costs in space, maintenance will mostly involve changing failed sub-assemblies rather than their repair. Teleoperators or robots may reduce the costs associated with extra-vehicular activity (EVA), but they must not be too specialized, because their capital costs should be shared by a wide variety of systems. This implies that coordination will be needed amongst PMSS system designers, so as to standardize maintenance procedures.

These considerations lead to several important guidelines for the PMSS power supply:

- The photovoltaic array should be modular, to simplify expansion to higher power, to facilitate maintenance, and to permit segmentation of commercial power. The individual modules should be small and readily replaceable, preferably using a simple teleoperator.
- The main PMSS power bus should be sized for much higher powers than the initial system can supply. Other distribution equipment should either be oversized or easy to replace.
- A separate, highly reliable, emergency power supply of modest output should be included in the system. This can enhance safety while reducing overall costs.
- In designing the main power plant, a principal objective should be cost minimization, commensurate with acceptable reliability. Minimizing mass and maximizing performance are secondary, less important objectives.

- For the main power plant, reliability assurance should be based on techniques used in terrestrial power systems rather than on extreme quality control and multiple redundancy.

It will not be easy to substitute commercial judgement for traditional priorities, especially in connection with issues related to the cost and reliability of equipment, but this change is essential to space commercialization. If NASA and the aerospace industry cannot demonstrate that these new requirements can be met, the PMSS will do more harm than good by suggesting that commercial space applications are not feasible.

THE ROLE OF GOVERNMENT

As presently planned, the PMSS will be a facility which is owned and operated by the U.S. Government. In particular, NASA will establish performance and design specifications for the power supply, although its construction will presumably be contracted to the private sector. NASA astronauts will erect and check out the power supply, and NASA will then sell electricity as needed by commercial users of the space station.

Other institutional arrangements are clearly possible. The power supply could be a government-owned, contractor-operated (GOCO) facility, or it could be developed, built and operated entirely by private industry. Indeed, an interesting near-term commercial market could be created in the provision of power (and other goods and services) to other users of space. A commercial organization, motivated to optimize profit rather than performance and reliability, may be able to build it at a significantly lower cost than in a Government procurement. Commercial space applications may provide corporate customers for this organization, but a solid market base exists already, including NASA, DOD, foreign government organizations, and the telecommunications industry. The space power company would have an interest in selling power to other users and could thus be expected to seek new space applications requiring electric energy on orbit.

A Federal commitment to the purchase, wherever possible, of products and services in orbit from commercial vendors would strongly encourage private investment in space systems. Appropriate measures to decrease downside risk (such as a guaranteed market) could result in growing traffic to orbit and thereby reduce launch costs for all users, including NASA and DOD. This approach would also reduce near-term pressures on the Federal space budget, by shifting some of the PMSS capital burden to the private sector.

Partial Federal assumption of the downside risk, by long-term contracts or other types of market guarantee, does not constitute a form of off-budget financing for government space systems. From a financing point of view, there is no more reason for NASA to own the PMSS power supply than there is for the agency to install generators in the basement of NASA HQ in Washington. Indeed, to the degree that facilities in orbit are intended to serve commercial interests, present policies represent a willingness by NASA to provide capital for the private sector.

In current jargon, space commercialization generally means the development of businesses which make use of the STS, the space station, or other space facilities, producing goods and services for use on Earth. The approach suggested here involves "privatization" of some of these facilities, which would otherwise be built and operated by NASA. The present policy of NASA is that privatization of elements of the PMSS is acceptable where the interface with the rest of the station is simple. For example, private development of free-flying experiment modules would be welcome. NASA also proposes to retain control of systems which are critical to mission success. According to these criteria, the power distribution system within the station and the emergency back-up power supply cannot be privatized, because these systems are essential to safety. If it is appropriately designed, the main, commercial power plant (including the solar array, energy storage, power conditioning, etc.) can readily meet the criteria for privatization: the interface may be no more complex than a power plug, and occasional black-outs are acceptable.

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1. Salkeld, R., et al: Space Transportation Systems, 1980-2000. Aerospace Assessment Series, Vol. 2, AIAA, NY 1978
2. Akin, D.L.: Optimization of Space Manufacturing Systems. In Grey, J., & Krop, C., Eds.: Space Manufacturing III, AIAA, NY, 1979
3. Energy in Transition, 1985-2010: Final Report of the Committee on Nuclear and Alternative Energy Systems, National Research Council, National Academy of Sciences. W.H. Freeman, San Francisco, 1979, p.368.
4. Chapman, P.K.: The Industrial Impact of the SPS. In Boeing Aerospace Co.: SPS System Definition Study, Final Report, Phase II, Vol. III, 1979 (NASA Contract #NAS9-15636)
5. Allen, C.W.: Astrophysical Quantities, 2nd Edition. The Athlone Press, University of London, London, 1962.
6. Van Nostrand's Scientific Encyclopedia, 5th Edition, Van Nostrand Reinhold Co., NY, 1976, p.1313.

TABLE I. - ENERGY COSTS FOR POTENTIAL INDUSTRIAL PROCESSES

Process	Specific Energy (kWh/kg)	Specific Energy Cost* (\$/kg)
Vaporization of water (Ref. 5)	0.63	63
Ionization of cesium (Ref. 5)	0.8	80
Electrolysis of water (Ref. 6)	6.2	620
Ionization of argon (Ref. 5)	10.5	1050
Dissociation of hydrogen (Ref. 5)	60	6000
Czochralski growth of Si boule (Ref. 4)	90	9000

*@ \$100/kWh

EVALUATING NASA TECHNOLOGY PROGRAMS IN TERMS OF PRIVATE SECTOR IMPACTS

Joel S. Greenberg
ECON, Inc.

NASA is currently developing spacecraft technology for application to NASA scientific missions, military missions and commercial missions which are part of or form the basis of private sector business ventures. The justification of R&D programs that lead to spacecraft technology improvements encompasses the establishment of the benefits in terms of improved scientific knowledge that may result from new and/or improved NASA science missions, improved cost effectiveness of NASA and DOD missions and new or improved services that may be offered by the private sector (for example communications satellite services). It is with the latter of these areas that attention will be focused upon. In particular, it is of interest to establish the economic value of spacecraft technology improvements to private sector communications satellite business ventures. It is proposed to assess the value of spacecraft technology improvements in terms of the changes in cash flow and present value of cash flows, that may result from the use of new and/or improved spacecraft technology for specific types of private sector communications satellite missions (for example domestic point-to-point communication or direct broadcasting). To accomplish this it is necessary to place the new and/or improved technology within typical business scenarios and estimate the impacts of technical performance upon business and financial performance. The ability to accomplish this has already been demonstrated* and is based upon the development

*Greenberg, J.S. and G.A. Hazelrigg, "Methodology for Reliability-Cost-Risk Analysis of Satellite Networks," J. of Spacecraft and Rockets, Vol. 11 No.9, September 1974.

Greenberg, J.S., "The Economic Implications of Unreliability," Proceedings of the Annual Reliability and Maintainability Symposium, January 1976.

Greenberg, J.S. and R. Nichols, "Economic Impact of New Technology on Domestic Satellite Communications," AMS Report No. 1285, Princeton University, March 31, 1976.

and modeling of business scenarios in a manner that converts changes in technology into changes in financial performance measures such as profit, cash flow, etc.

The economic value of spacecraft technology improvements (such as ion thrusters and batteries) can be assessed in terms of changes in cash flows and their present values associated with typical domestic point-to-point communications satellite and direct broadcast satellite missions. Projections can then be made of the number of such missions as a function of time and the aggregate benefits (changes in private sector communication satellite business venture cash flows and present values) estimated. It is also possible to extend these results so as to yield estimates of the impact of likely foreign technology advances relative to U.S. technology advances through estimates of the impacts on imports/exports by capturing of the spacecraft markets as a result of technology differences.

In order to accomplish the above it is necessary to formulate typical communications satellite business ventures, such as point-to-point communications and direct broadcast. The formulation and structuring of the business ventures includes the characterization of the market (i.e., demand in terms of protected and non-protected transponders), sparring philosophy, use of insurance, methods of depreciation, policy with respect to write-off of premature failures, etc. As a result of the formulation of the business scenarios, typical sets of performance characteristics (i.e., channel capacity, power, reliability, number of beams, pointability, stability, etc.) can be established. Spacecraft can then be configured with and without the technology resulting from the NASA spacecraft technology programs. The performance attributes of these spacecraft in terms of subsystem life characteristics, mass, recurring and non-recurring costs can then be utilized to perform a financial analysis of the business ventures using these spacecraft.

The consequences of utilizing the different spacecraft configurations (resulting from different levels of technology resulting from the NASA programs) in the typical business ventures can be accomplished by using a financial simulation model that takes into account (explicitly and quantitatively) uncertainty in costs and performance and unreliability. The DOMSAT Model has such capabilities.

Greenberg, J.S., "A Benefit-Cost Analysis of Nuclear Power Applied to the GPS Mission", J. of Energy, September 1977.

Greenberg, J.S., "Evaluating the Economic Impact of Design Alternatives on Domestic Communication Satellite Ventures," ACTA Astronautica, December 1979.

The purpose of the DOMSAT type of model is to provide an economic measure of the value of introducing new or improved technology into the domestic communication satellite mission. The DOMSAT Model and associated program are probabilistic (Monte Carlo) so that the consequences of the explicit consideration of unreliability and system and cost uncertainties can be evaluated. It must be noted that because different technologies are in different stages of research, development, design and use, different levels of performance and cost uncertainties may exist and different levels of reliability may be anticipated. Data are thus specified as bounded probability distributions which represent subjective estimates of the possible values of pertinent parameters. The mathematical model represents a generalized domestic satellite communications mission under conditions of uncertainty and provides output information as probability distributions, expected values and standard deviations which reflect the uncertainty in the input data and the impact of unreliability. The program consists of:

- An operational section which simulates and records the performance and operational events such as system failures, launch attempts, satellites employed and communications system performance. The impact of using alternative systems and technologies, for example the satellite power subsystem, is registered through its effect on the simulated operation of the communication service.
- A financial section which establishes the annual revenues, expenses, profit, cash flow, etc., resulting from communication service.
- A market section which simulates the market environment surrounding the communication services and contains the decision processes which dictate the response of the communication service operator to the market model. The communications marketplace is considered to be a known through probabilistic function of time consisting of a mix of guaranteed (guaranteed in the sense that a contract exists which guarantees the availability of the channels) and nonguaranteed channels.

The output financial information is presented in the form of probability distributions describing quantities such as annual revenue, after-tax profit, cash flow, indebtedness and present value of cash flow. Additional information, such as certain expense items, are presented as expected annual values. Quantities of interest relating to the operational aspect, such as the number of launch attempts, number of satellites required, number of propulsion systems and satellites refurbished, are also available in the form of probability distributions.

The program provides a mechanism for establishing the value of technology and operational alternatives. The impact of both launch system and satellite technology changes can be observed and evaluated in terms of financial

measures such as annual profit, cash flow, net present value, etc. The following provisions are included in the model:

- Specification of the launch system to be used including the price of the service as a function of time and the technologies employed (type of orbit injection system used and ability to refurbish)
- Consideration of reliability of the launch system at the major subsystem level
- Spacecraft failure model which allows for initial, random and wearout failures
- Repeater redundancy between satellites based upon frequency-wise corresponding repeaters on separate satellites to provide a mutual backup facility
- Consideration of demand for communications over the time period of concern in the form of an annual demand input
- Incorporation of decision rules and threshold criteria which dictate the response of the communication system to the demand function. Of particular importance is the decision to initiate launching additional satellites to maintain the service.

The program determines the probability distributions of:

- Annual revenue
- Annual profit (loss)
- Annual cash flow
- Quantities pertinent to the service operations such as number of launch attempts, number of satellites purchased, number of propulsion modules refurbished and others
- Present value at several different discount rates.

From the financial results generated by a model such as described above, the impacts of technology improvements can be assessed in terms of the changes in cash flow, net present value, return on investment and risk. Judgements can then be made with respect to the desirability of the technology programs from the private sectors point of view—these judgements can in fact be quantified in terms of the impact on the net present value of cash flow.

This type of analysis can be expanded to include the impact of foreign spacecraft technology development programs. To accomplish this it is necessary to make projections of U.S. and foreign spacecraft technologies with and without NASA spacecraft technology programs. These projections must again be converted into spacecraft configurations which then serve as the basis of a database provided

to the DOMSAT type of financial model. Again, the impact of the spacecraft technology made possible by the technology development programs can be assessed in terms of the business venture's financial performance measures. Based upon the difference in the financial performance measures resulting from the use of foreign and U.S. technology based spacecraft, estimates may be made of the likelihood of the business ventures using "foreign" spacecraft. These estimates coupled with projections of the demand for communications satellites can lead to estimates of the potential impact of foreign and U.S. spacecraft technology development programs on U.S. imports and exports.

MISSIONS/PLANNING PANEL

J. Stuart Fordyce, Moderator
NASA Lewis Research Center

A panel discussion was held to develop a viewpoint on space power technology needs and state of readiness for future mission scenarios. The panelists - Cosmo Baraona, Space Station Task Force, NASA Headquarters; Richard A. Wallace, SP-100, Manager, Mission Requirements Analysis, JPL; Jesco von Puttkamer, Program Manager, Long Range Planning, OSF, NASA Headquarters; William L. Piotrowski, Manager, Planning Office, OSSA, NASA Headquarters; and Dr. Richard J. Williams, Deputy Chief, Solar System Exploration Division, JSC and leader of the lunar initiative, were asked to summarize their reactions to eight questions shown on Figure 1 to initiate discussion.

Among the points made in the discussion, it was agreed that missions, particularly the far term ones, do serve to drive technology; however, as the missions become nearer term, issues of schedule and cost severely limit the willingness to accept risk. There are, in fact, no rewards to a mission manager for introducing new technology. Mission downscaling is the usual response to technology limitations. All panelists agreed that there exists a serious gap between when technologists feel their job is done and what mission managers need for decision. Typically a two to three year engineering development gap exists. It is essential to take technologies to the engineering model level and conduct a flight demonstration to close this gap. All agreed that increased effort should be made to achieve stronger interactions between planners and technologists and that workshops like the present one are a step in the right direction. Technologists need mission credibility and vice versa.

STARTERS

1. BASED ON THE MISSION OVERVIEWS PRESENTED, WHAT MISSIONS CANNOT BE UNDERTAKEN BASED ON THE PRESENT SOA IN POWER TECHNOLOGY? WHERE IS THE GREATEST LEVERAGE?
2. WHAT DO YOU DEFINE AS AN "ENHANCING"/"ENABLING" TECHNOLOGY?
3. WHAT ARE THE TECHNOLOGY CRITERIA YOU USE IN PLANNING A MISSION?
4. HOW DO YOU ASSESS THE RISK OF USING A NEW TECHNOLOGY IN PLANNING A MISSION?
5. DO YOU BELIEVE THAT MISSIONS SHOULD SERVE AS TECHNOLOGY DRIVERS OR SHOULD MISSION PLANNERS CONSTRAIN THEIR PLANNING TO THE USE OF CURRENT SOA TECHNOLOGY?
6. IS THERE SUFFICIENT INTERACTION BETWEEN THE TECHNOLOGISTS AND THE MISSION PLANNERS? IF NOT, HOW CAN WE IMPROVE THE CONNECTION BETWEEN THE ADVANCING TECHNOLOGY AND MISSION NEEDS?
7. HOW MUCH CONSIDERATION IS OR SHOULD BE GIVEN TO PROVIDING FOR REPAIR OR REPLACEMENT OF POWER SYSTEMS OR SUBSYSTEMS, I.E., CAN WE REDUCE COST, WEIGHT, ETC. BY UTILIZING MANNED INTERVENTION IN LIEU OF "FAILSAFE" DESIGNS? TO WHAT EXTENT SHOULD THIS BE PRACTICED IF AT ALL?
8. ARE THE TECHNOLOGY ISSUES DIFFERENT FOR THE MILITARY, CIVIL AND COMMERCIAL MISSIONS? HOW SO? ARE YOU OPTIMISTIC ABOUT PRIVATE INVESTMENT PROVIDING SPACE CAPABILITY IN THE FUTURE?

omit
70
P.75

NASA SPACE ENERGY SYSTEMS PROGRAMS

J. P. Mullin
NASA Headquarters

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ENERGY SYSTEMS COMPRISE MAJOR MASS

(EXCLUDING PAYLOAD)

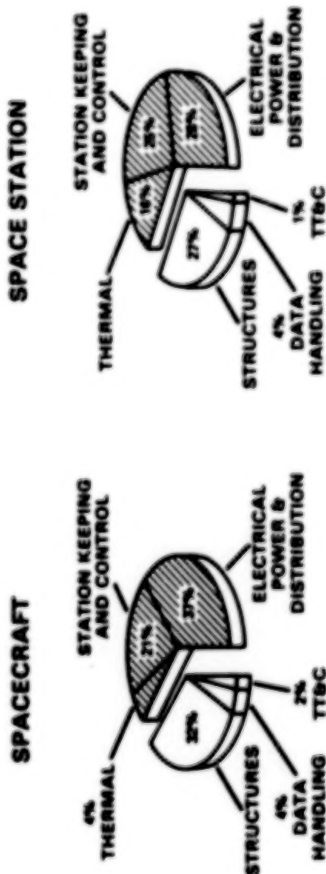


Figure 1.

PHOTOVOLTAICS

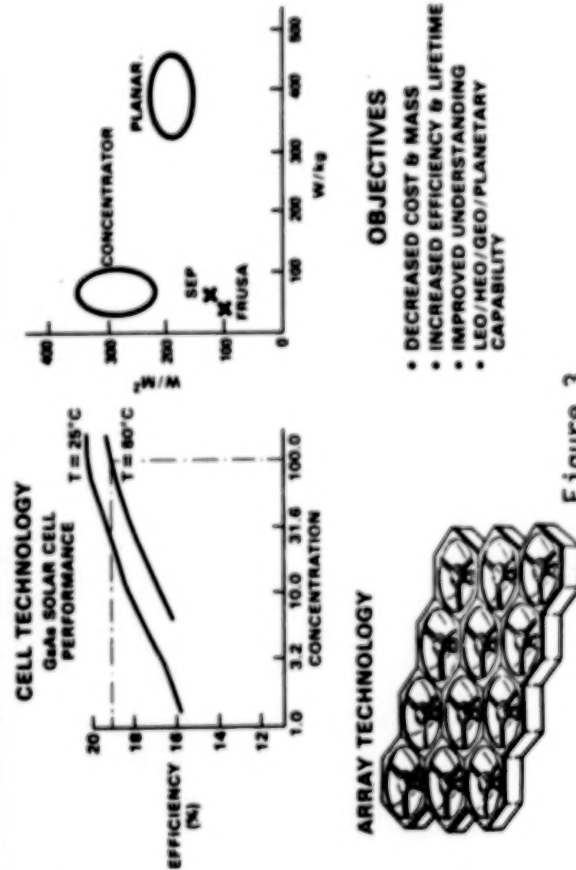


Figure 3.

MAJOR PROGRAM THRUSTS

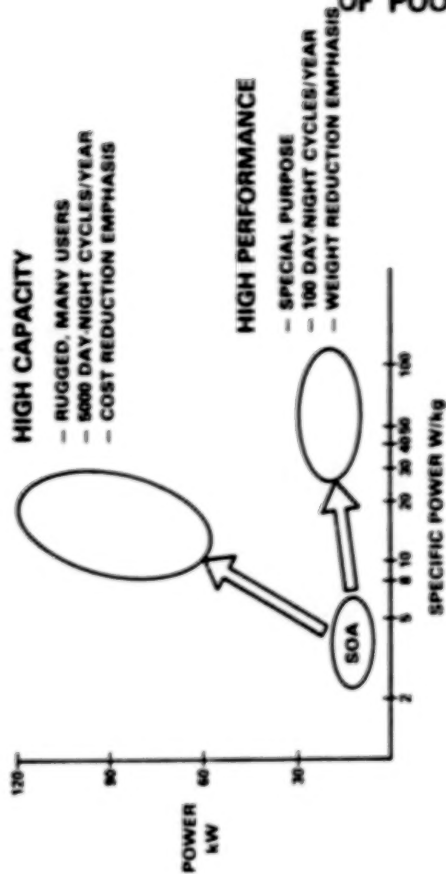


Figure 2.

PHOTOVOLTAIC ENERGY CONVERSION

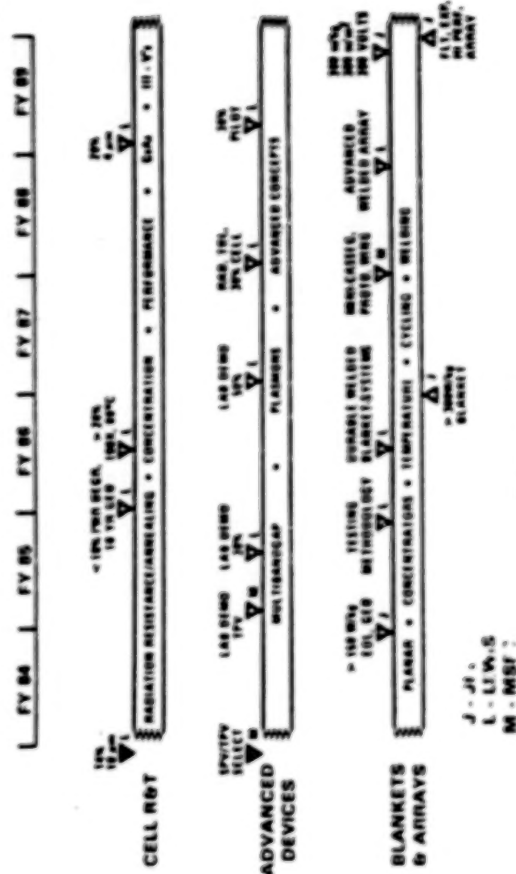


Figure 4.

LIGHTWEIGHT ARRAYS FOR SPACECRAFT

R&T ADVANCES

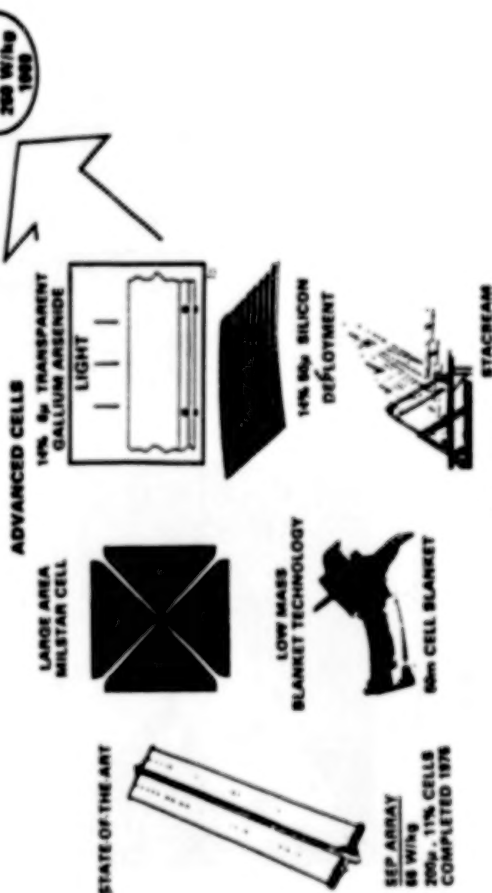


Figure 5.

PRIMARY LITHIUM BATTERY R&T

OBJECTIVES

- 10x INCREASE IN ENERGY DENSITY
- UNDERSTAND CHEMISTRY
- ESTABLISH CONTROLLED FABRICATION PROCESSES



ACCOMPLISHMENTS

- HAZARDOUS INTERMEDIATES IDENTIFIED
- HEAT GENERATION MODEL DEVELOPED

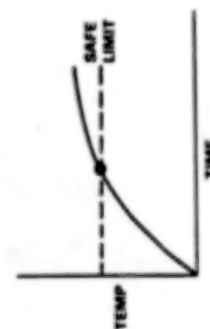
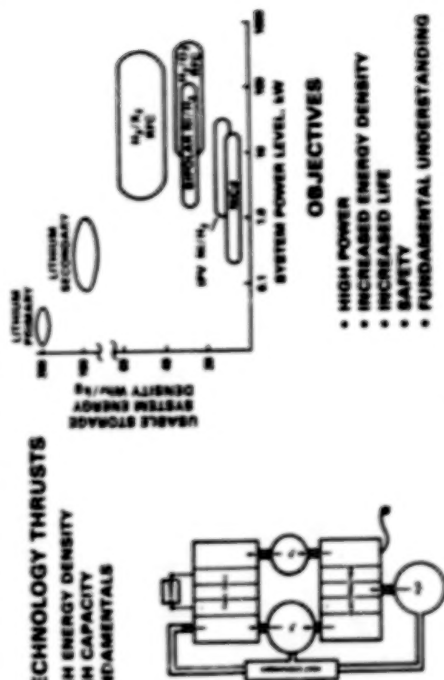


Figure 7.

CHEMICAL ENERGY CONVERSION AND STORAGE

TECHNOLOGY THRUSTS

- HIGH ENERGY DENSITY
- HIGH CAPACITY
- FUNDAMENTALS

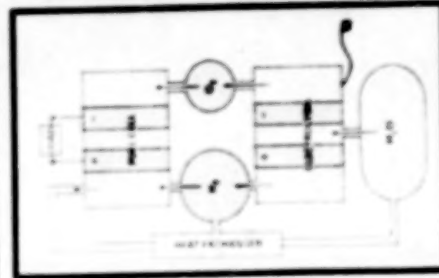


OBJECTIVES

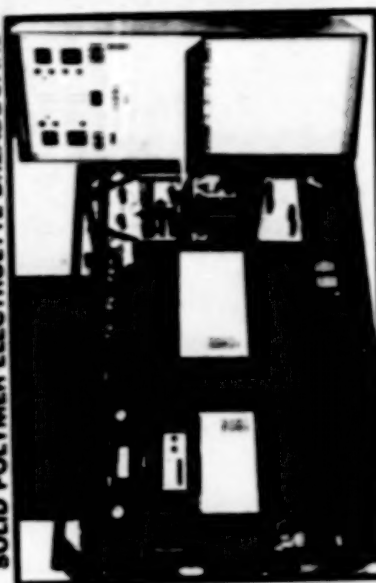
- HIGH POWER
- INCREASED ENERGY DENSITY
- INCREASED LIFE
- SAFETY
- FUNDAMENTAL UNDERSTANDING

Figure 6.

TECHNOLOGY FOR HIGH CAPACITY ENERGY STORAGE SYSTEMS



SOLID POLYMER ELECTROLYTE BREADBOARD



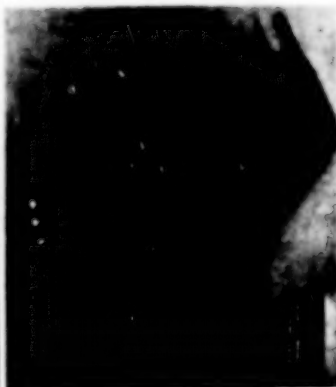
BREAD BOARD EVALUATION-LEO REGIME

Figure 8.

BIPOLAR NICKEL-HYDROGEN BATTERY R&T

- > 2000 CYCLES
- 1 KW PEAK POWER

ENGINEERING
MODEL
1986



- HIGH PRECISION CONSTRUCTIBILITY
- ACTIVE COOLING
- COMPACT DESIGN

Figure 9.

PLASMA INTERACTIONS EXPERIMENT



PLASMA INTERACTIONS EXPERIMENT

- 2000cm² SOLAR ARRAY
- BIAS \pm 1000V
- MEASURE COLLECTED CURRENTS AND BREAKDOWNS

- COMPARE DATA WITH PREDICTIONS

- SECONDARY DELTA PAYLOAD
- LAUNCH SCHEDULED AUG. 1982



SOLAR ARRAY MODULE

Figure 11.



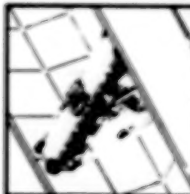
Figure 10.

VOLT

VOLTAGE OPERATING LIMIT TESTS

FOUR STS FLIGHT EXPERIMENTS

- VOLT-1: 1985**
- BIASED PLANAR ARRAY SEGMENTS (± 1000 VOLTS)



- VOLT-2: 1987 (EARLY)**
- PLANAR ARRAY
 - SELF-GENERATED VOLTAGES TO ~ 800 VOLTS



- VOLT-3: 1988**
- BIASED CONCENTRATOR ARRAY SEGMENTS (± 1000 VOLTS)



- VOLT-4: 1987 (LATE)**
- CONCENTRATOR ARRAY
 - SELF-GENERATED HIGHER VOLTAGES



Figure 12.

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ENABLING SPACE MISSIONS BY LASER POWER TRANSMISSION

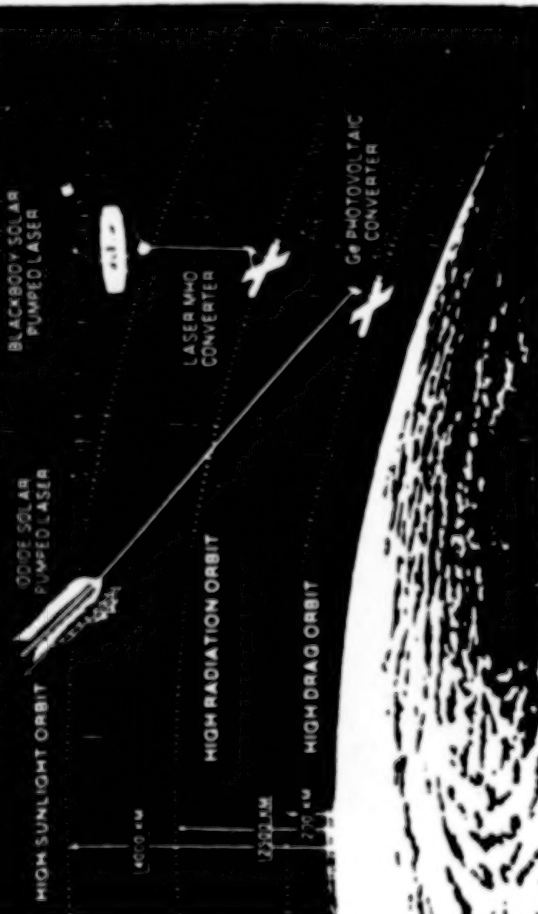


Figure 13.

TWO PHASE THERMAL MANAGEMENT

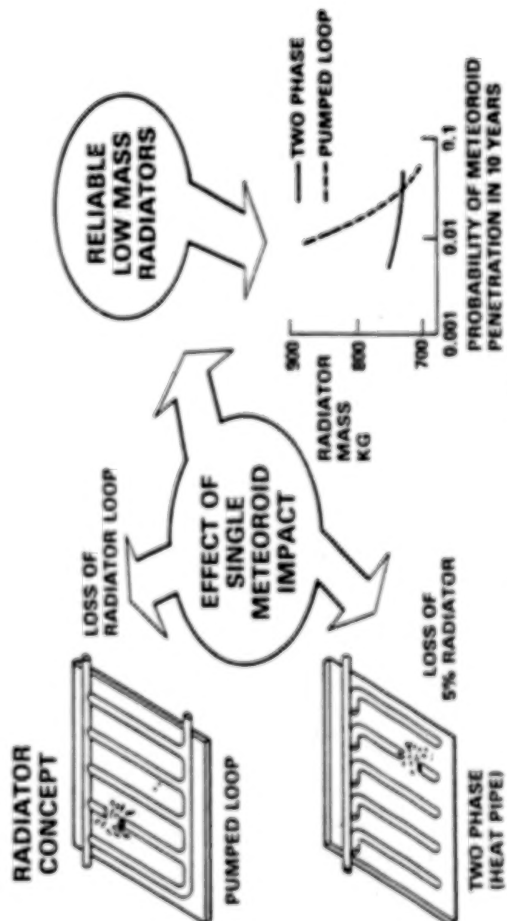


Figure 15. • 32 KW

TWO PHASE THERMAL MANAGEMENT

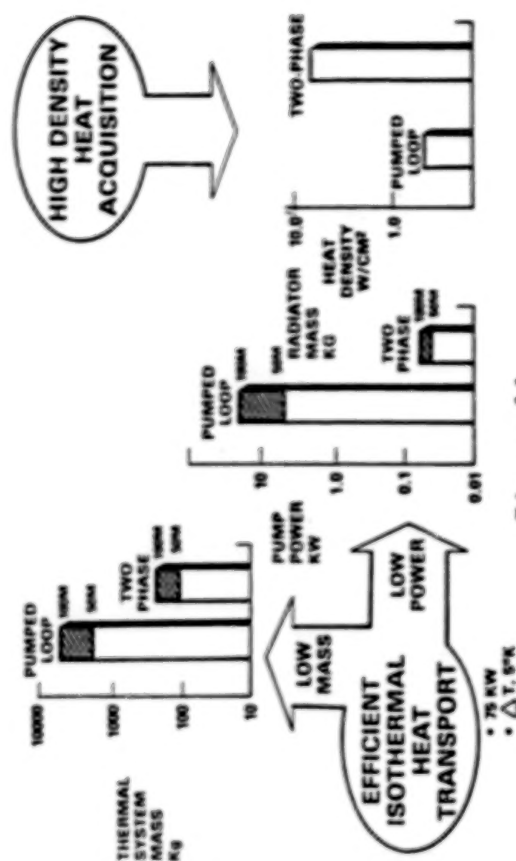


Figure 14.

50-FOOT LONG PROTOTYPE RADIATOR HEAT PIPE WITH MULTILEG EVAPORATOR

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Figure 16.

LIQUID DROPLET RADIATOR CONCEPT

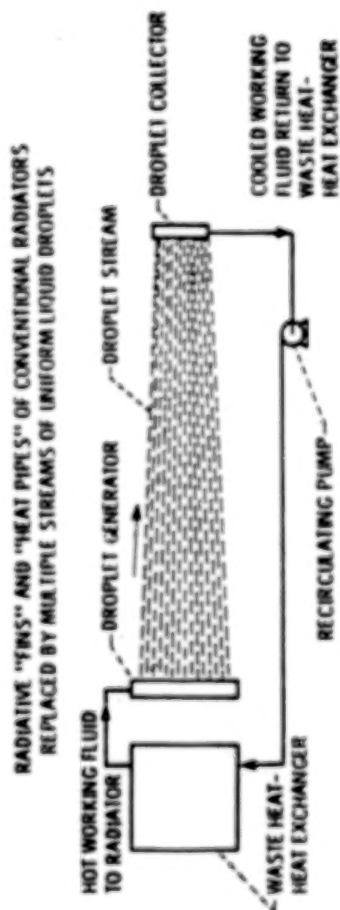
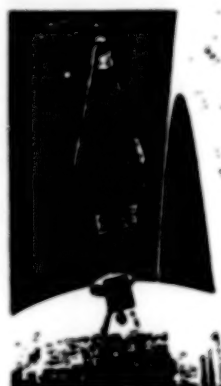
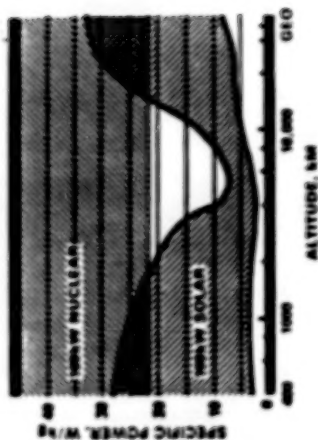


Figure 17.

THERMAL-TO-ELECTRIC CONVERSION

- TECHNOLOGY THRUSTS**
- CONVERSION TECHNOLOGIES
 - SYSTEM DESIGN & ANALYSIS
 - MISSION DEFINITION



OBJECTIVES

- DECREASED COST
- INCREASED SYSTEM SPECIFIC POWER
- HIGH POWER/LONG LIFE
- SAFETY

Figure 18.

TECHNOLOGY DEVELOPMENT

- ADVANCED MATERIALS**
- REFRACTORIES
 - THERMOELECTRICS
- CONVERSION**
- THERMOELECTRICS
 - STIRLING

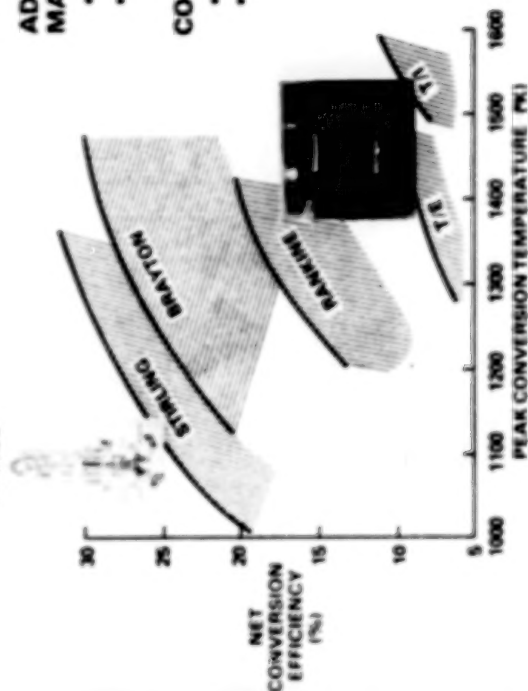


Figure 20.

CIVIL MISSIONS ENABLED OR ENHANCED BY NUCLEAR REACTOR POWER

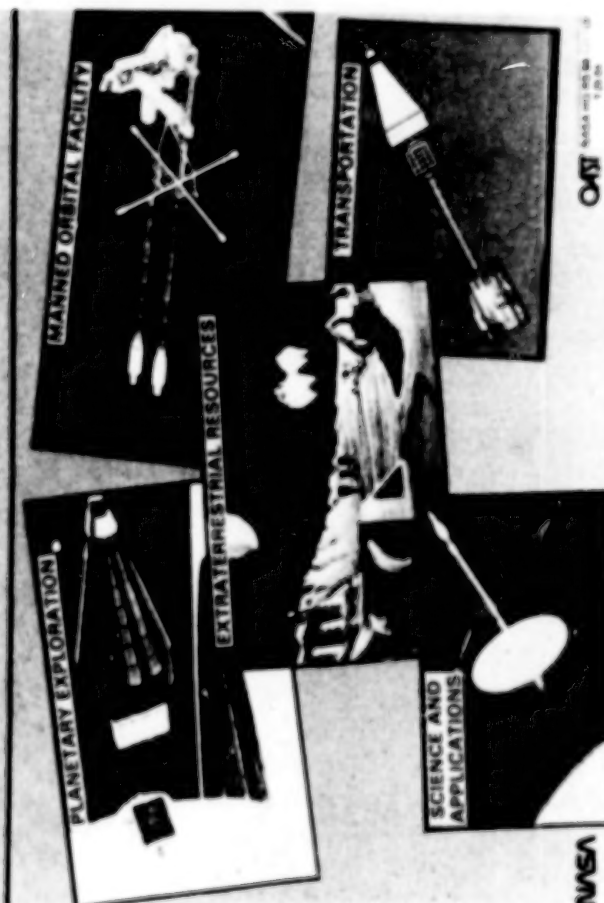


Figure 19.

SPACE ENERGY SYSTEMS



Figure 21.

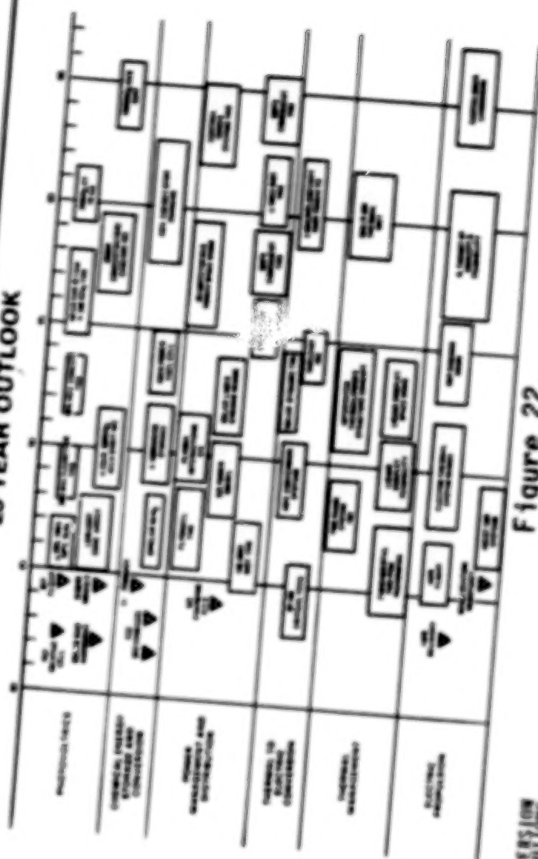


Figure 22.

SPACE ENERGY CONVERSION
SOLAR ENERGY CONVERSION

- 0 SOLAR CELL RADIATION HANDBOOK
- 0 HIGH POWER/VOLTAGE TRANSISTORS
- 0 LARGE AREA WRAPAROUND SOLAR CELLS
- 0 SPACECRAFT CHARGING MODELS
- 0 POLYPROPYLENE BATTERY SEPARATOR
- 0 ION ACCELERATOR TECHNOLOGY
- 0 PEP REFIN ADHESIVE
- 0 EVALUATION OF SELENIDES FOR RTG
- 0 HOLLOW CATHODE PLASMA SOURCES
- 0 BACK SURFACE FIELD AND REFLECTOR
- 0 FUEL CELL TECHNOLOGY
- 0 HIGH POWER CONCEPTS (SOLAR ARRAY, FUEL CELL, ELECTRONICS, THERMAL, AND PROPULSION)

COMMENT/USE/APPLICATION

INTERNATIONAL STANDARD REFERENCE

AMERICAN IN OILS ALTHOUGH PRODUCTS ARE

THE NEW FORCE MILSTAR PROGRAM

STANDARD FORMS (TOMS/GALILEO/MARCS)

CONFIDENTIAL PRODUCT! BROADLY USED

ADDITIONAL INFORMATION ON THE STRUCTURE OF THE

THE UNIVERSITY OF TEXAS AT AUSTIN

NEW AIR FORCE NEWS SOURCES

CONSUMPTION FUN SPACECRAFT CHARGE

5700 WATTS MONITOR FOR SILICON SOLAR CELLS

Billions of dollars in new

IN THE FLIP/POWER MODULE AND SPACE STATION RESEARCH

1000

- 0 LAUNCHED/DIRECTED HIGH CAPACITY
POWER AND THERMAL R&D
- 9 INITIATED/LED ADVOCACY FOR SP-100
- LED DEVELOPMENT OF KEY PLP. INITIATIVES
SAGE VOL
TAPS
TFAP

⁸ FORCES FOR NATIONAL SPACE ENERGY EFFORT

Figure 23.

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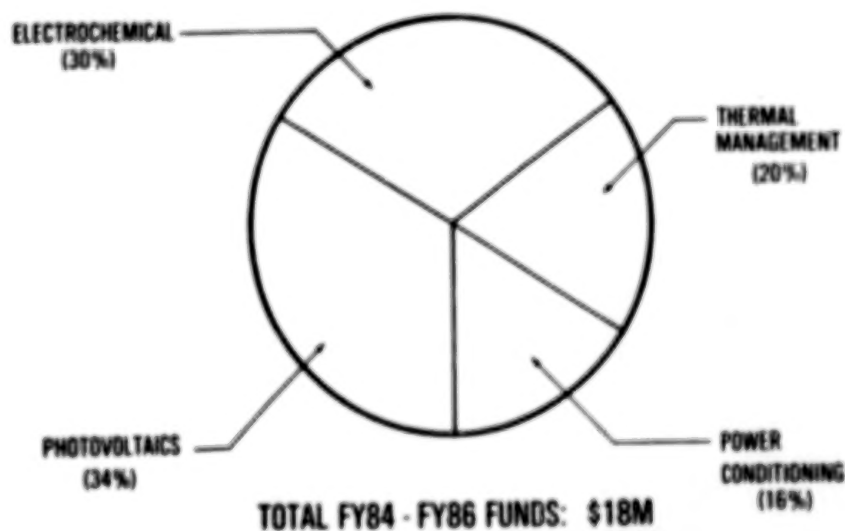
AIR FORCE SPACE POWER TECHNOLOGIES

J. D. Reams
Air Force Wright Aeronautical Laboratories
Wright-Patterson Air Force Base

No text available at time of printing.

- PHOTOVOLTAICS
- ELECTROCHEMICAL POWER SOURCES
- THERMAL MANAGEMENT
- POWER CONDITIONING

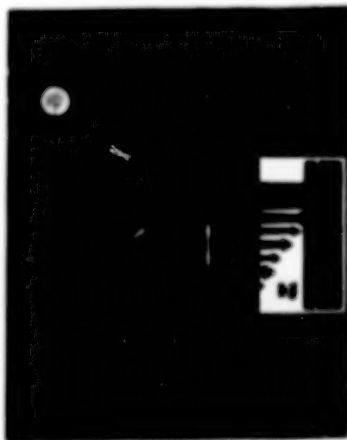
Air Force Space Power Technologies to be covered in this overview.



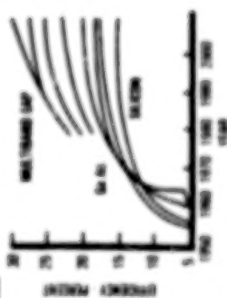
The research and development investment devoted to these Space Power Technologies over the FY84 through FY86 time period is shown in the pie chart. The investment in thermal management and power processing may increase significantly as Strategic Defense Initiative SDI funding becomes available.

THE TECHNOLOGIES

- SILICON CELLS
- GALLIUM ARSENIDE (GaAs) CELLS
- MULTI-JUNCTION (MJ) CELLS



STATUS



THE GOALS

- EFFICIENCY
- LIGHT WEIGHT
- RADIATION HARDENING
- SURVIVABILITY

Although there is still some interest in thin silicon technology, most of the cell work is with GaAs in single and multi-junction versions. The projected efficiency chart clearly justifies that focus. A major goal of cell development is survivability to laser and nuclear threats with a minimum penalty in performance. High end of life efficiency and light weight will continue to be major development goals also.

VAPOR DEPOSIT GALLIUM ARSENIDE



VAPOR DEPOSIT GERMANIUM



PREPARE FOR PROCESSING



LOW MASS / LOW COST SUBSTRATES-GALICON

SUBSTRATE TYPE	GALECON	GAAS
MASS DENSITY	14.33 MG/CM	108.71 MG/CM
COST PER AREA	\$5.50/CM	\$4.00/CM

* 200 MICRONS THICK - TENSILE MEMBERS
AND DIFFICULT TO HANDLE

This chart describes the silicon substrate application to GaAs cell technology.

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GOALS	FY83	FY84	FY85	FY86	FY87	FY88
EFFICIENCY						
LIGHTWEIGHT						
RADIATION HARDENING						
SURVIVABILITY						

In addition to basic cell technology development, other programs are investigating: (a) the improvement of GaAs cell packaging using silicon substrates for higher strength of thin cells, (b) integral covering of large areas from a plasma activated source, and (c) high temperature contacting which permits short term temperature exposures up to 500°C. Radiation degradation tests are ongoing at JPL (proton) and at NBS (neutroning of radiation damage versus temperature).



BUS BAR
CELL
SUBSTRATE
COVER

LASER IRRADIATED SOLAR CELLS WITH ADHESIVELY BONDED COVERS



Laser irradiation threats present a serious challenge to cell designs. High temperature contact, reflective coatings, and integral covers play a role in hardening to laser threats.

THE TECHNOLOGIES

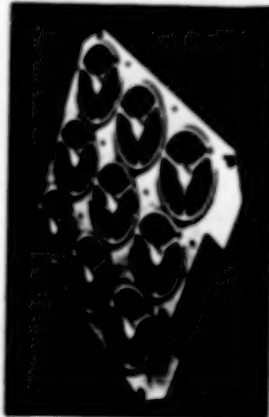
- PLASMA ARRAYS
- CONCENTRATOR ARRAYS

STATUS

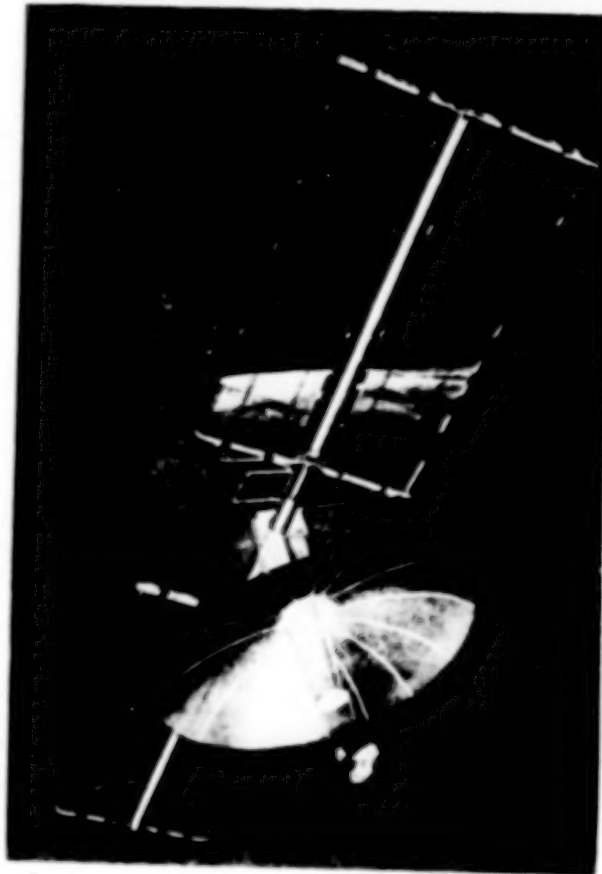


THE GOALS

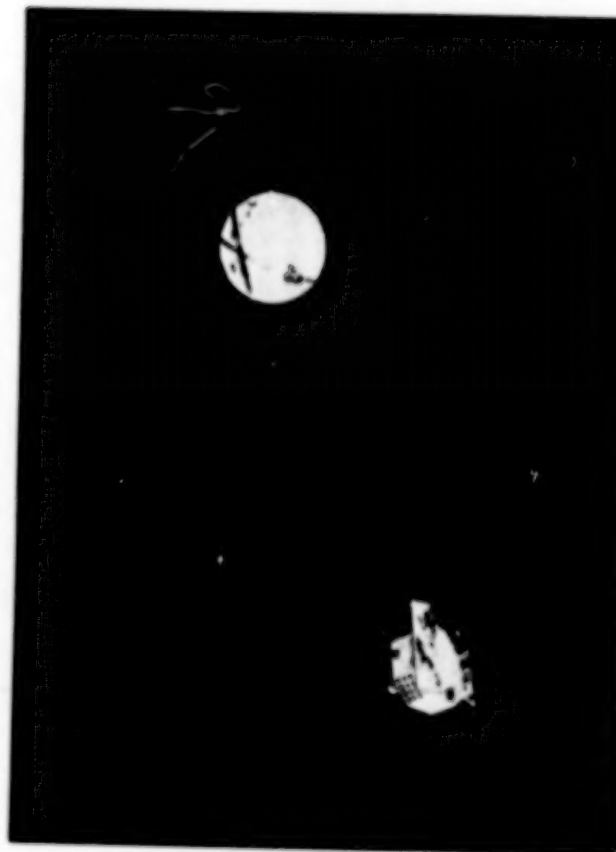
- LIGHT WEIGHT
- HIGHER POWER LEVEL
- SURVIVABILITY



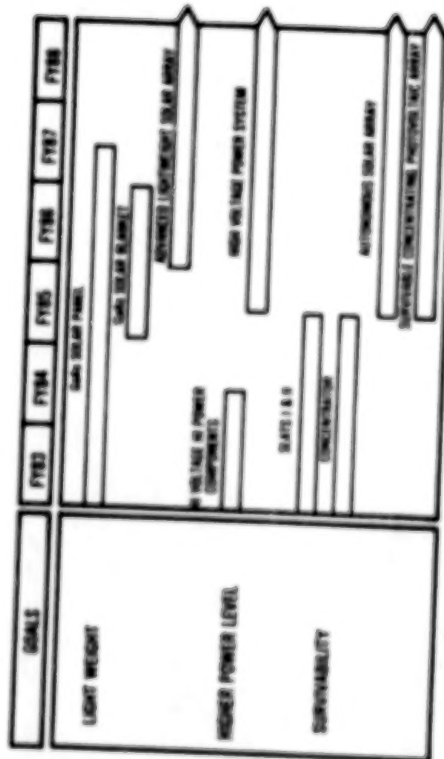
The Solar Array program emphasizes concentrator technology with some effort given to plasma arrays. Survivability at various laser threat levels may be possible only with concentrator systems. However, lightweight plasma arrays will be needed where survivability is not a major requirement.



This chart shows an artist's concept of the High Voltage Power System. Advanced cell technology, advanced structures, high voltage distribution, advanced power processing, and high energy density battery technology would jointly provide 10 to 12 watts per pound of solar array/battery power system.

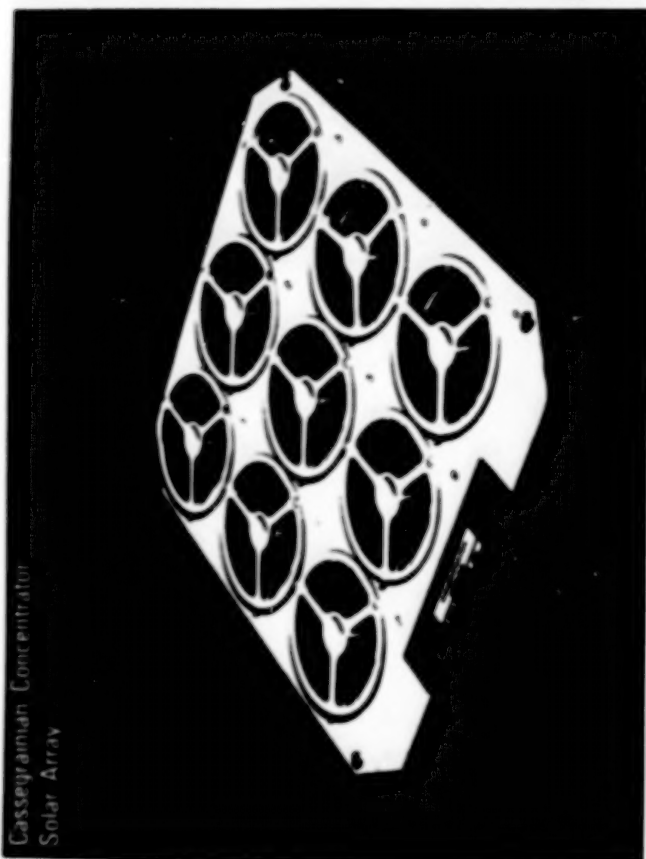


Reflected degradation of Gals cells on real time space tests is planned on the Combined Radiation Release Experiment Satellite (CRRS). This vehicle is expected to fly in a highly elliptical orbit from near GEO around to low earth perigee, thereby crossing strong radiation belts many times during the more than 2 years flight duration.



A Gals panel effort has provided small space experiment samples for several piggyback flights such as the Navy's Living Flame Shield Space Experiment. The effort is also providing the CRRS panel which is expected to fly in 1987. Other outerplaner array program do not have funds programmed. If funded, the high voltage power system array would flight qualify an advanced technology array using advanced cells, high voltage distribution, advanced structures, and power processing.

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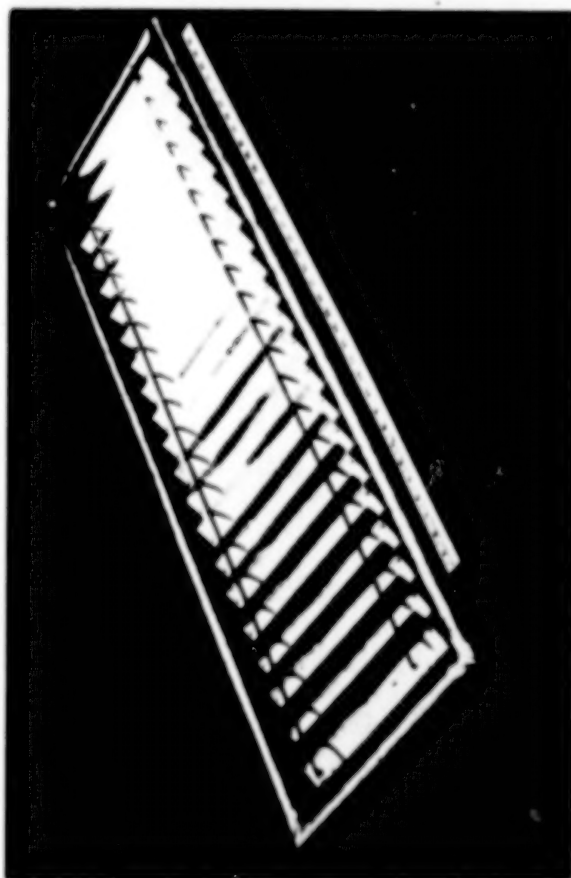


Cassegrain Concentrator
Solar Array

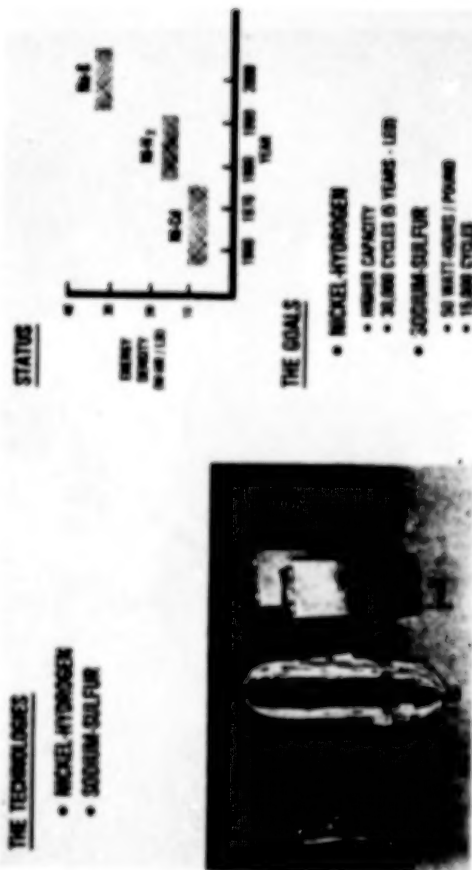
The Cassegrain concentrator concept has been funded by the Air Force and NASA with different objectives. NASA's objective being low cost and the Air Force's primary objective is hardness to laser irradiation.



This chart depicts the growth in energy density projected for NaS battery technology.



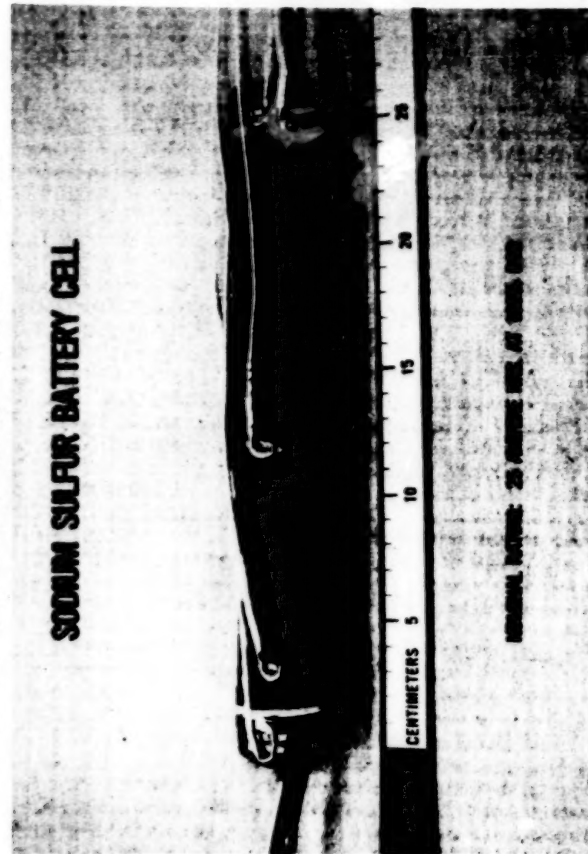
This chart shows a small model of the Survivable Low Aperture Trough Concentrating Array. This development effort is jointly funded by the Air Force, Navy, and NASA.



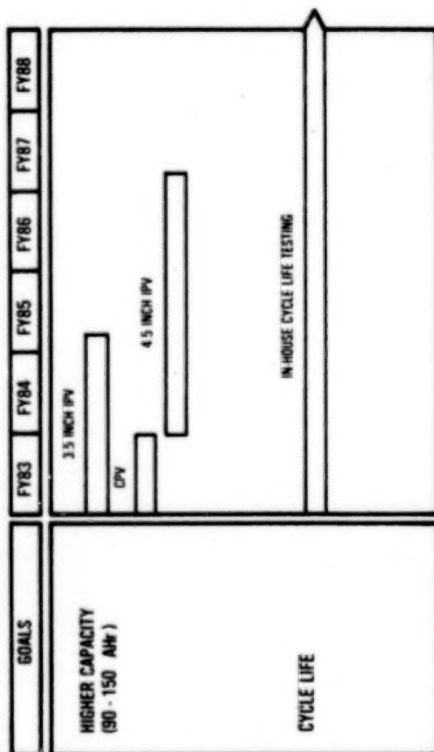
Advanced battery technology includes large size (4.5 inch) nickel-hydrogen cells and sodium-sulfur cells. Increased life and usable energy density are primary objectives of advanced cell designs.



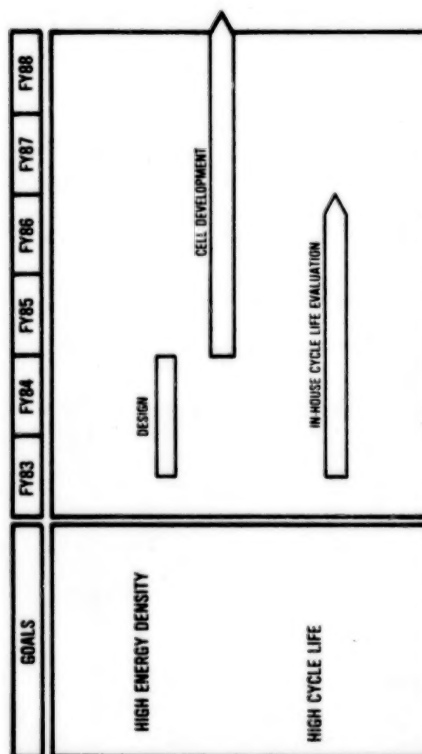
This chart depicts one-half of a battery for the Satellite Data System.



This is an experimental NaS cell.



The Ni-Hg program has recently been redirected from the common pressure vessel (CPV) concept to a larger size (4.5 inch) 90 - 150 ampere-hour capability. Life cycle tests are a current part of the development effort.

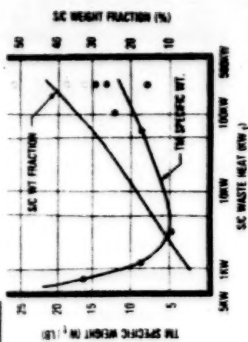


The NaS cell development program is focused toward a battery energy density of 40 or more watt-hours per pound and up to 15,000 cycles.

STATUS

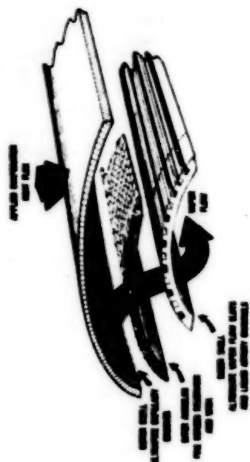
- PUMP AUGMENTED HEAT PIPES
- TWO-PHASE CO-CURRENT, LOOPS
- INTEGRAL PULSED POWER THERMAL STORAGE

DOUBLE WALL HEAT PIPE STRUCTURE DETAIL

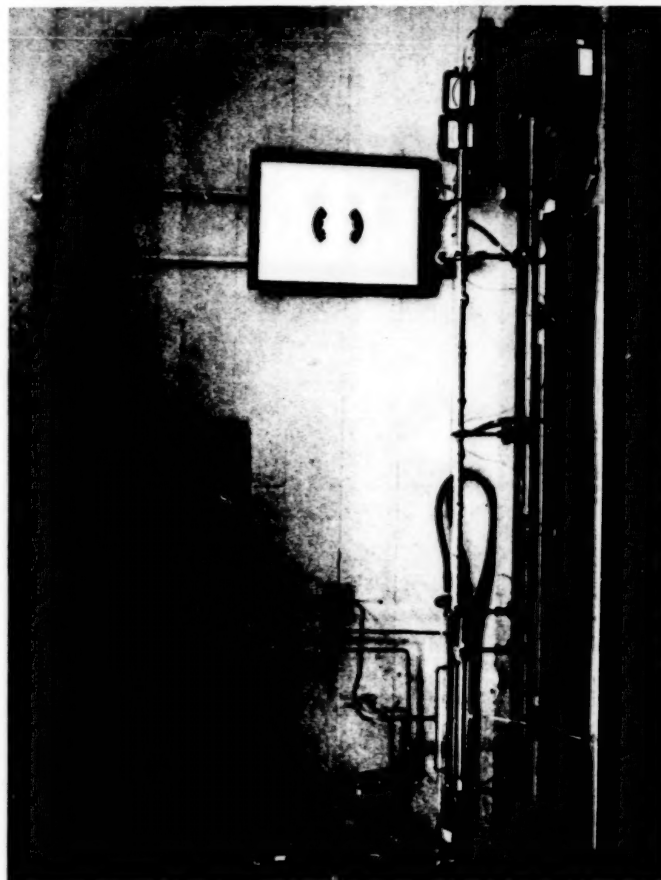


THE GOALS

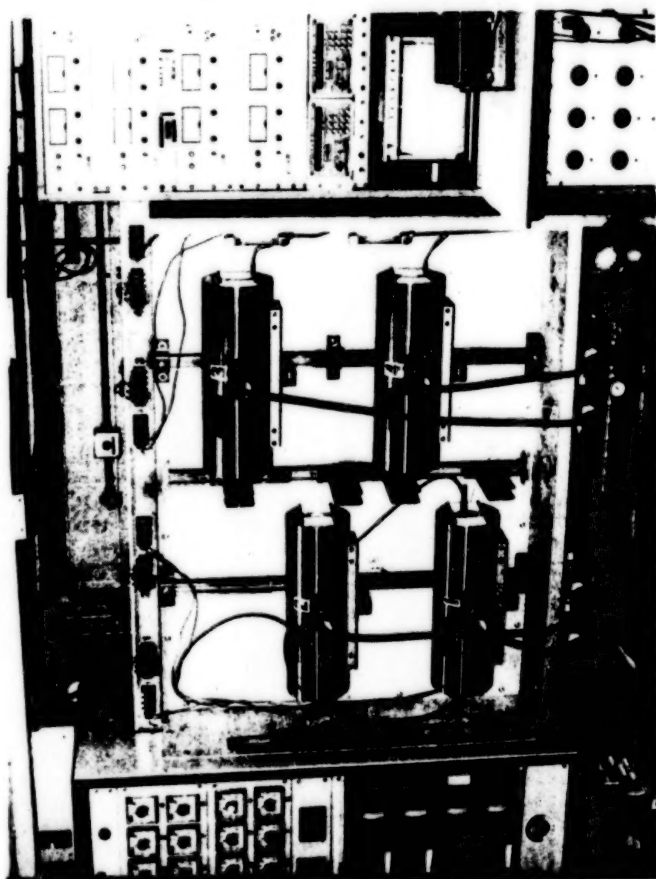
- PERFORMANCE IMPROVEMENT
- 100X TRANSPORT CAPACITY IMPROVEMENT
- 50% REDUCTION IN MASS, PARASITIC POWER
- UP TO 100%¹ PEAK / AVERAGE WASTE HEAT TRANSPORT



The thermal management program includes advanced heat transport devices and advanced heat rejection technology which will be discussed later. Advanced heat pipes capable of 100% greater transport capacity are needed for high power systems. Peak to average ratios of 100 for waste heat transport are required for pulsed power systems.



This picture shows an in-house test of a double wall heat pipe. 2.0 kilowatt-meters heat transport capacity has been demonstrated with a 1.0 meter pipe. A new 2.0 meter pipe (just starting tests) has a goal of 10.0 kilowatt-meters.



This picture shows 4 NaS cells in insulated containers on test in the Aero Propulsion Laboratory. These are 25 amp-hour experimental cells of the type shown on the previous chart.

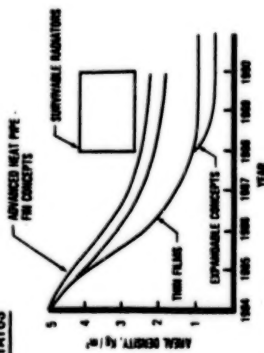
GOALS	FY83	FY84	FY85	FY86	FY87	FY88
PERFORMANCE IMPROVEMENTS						
PEAK / AVERAGE WASTE HEAT TRANSPORT						

Performance improvement efforts in heat pipe designs include advanced configurations and pump augmentation. The peak to average burst power thermal management is receiving increased emphasis.

THE TECHNOLOGIES

- HIGH CAPACITY, HEAT PIPE / COMPOSITE FIN RADIATORS
- THIN FILM CONFINED / UNCONFINED RADIATORS
- EXPANDABLE RADIATORS
- SURVIVABLE RADIATORS

STATUS



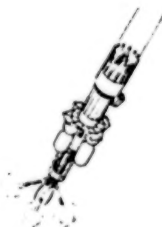
THE GOALS

- 50% REDUCTION OF SPECIFIC MASS
- UP TO 1000/1 PEAK / AVERAGE HEAT DISSIPATION
- SURVIVABLE TO LASER THREAT

Advanced radiator technology promises significant specific mass reductions. High peak to average (1000 to 1) heat dissipation and survivability to laser threats are objectives of the effort.



VARIABLE CONDUCTANCE
HEAT PIPE RADIATOR



EFFLUENT THERMAL CONTROL



EXPANDABLE RADIATOR

This chart depicts several means for acquiring pulsed power heat rejection.

GOALS	FY83	FY84	FY85	FY86	FY87	FY88
REDUCTION IN SPECIFIC MASS						
1000/1 PEAK / AVERAGE HEAT DISSIPATION						
SURVIVABILITY						

THIN FILM RADIATOR TECHNOLOGY

EXPANDABLE RADIATOR TECHNOLOGY

SURVIVABLE NUCLEAR SIC THERMAL

HEAT PIPE / COMPOSITE FIN RADIATOR

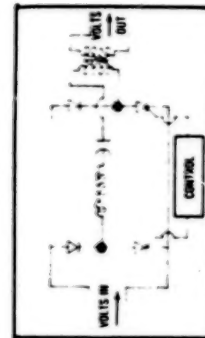
IN HEAT FLUX ELEC. RAD

The funded program includes thin film confined and unconfined radiator technology and expandable radiator concepts. Advanced concepts for advanced batteries and high heat flux electronics are projected for the future.

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THE TECHNOLOGIES

- HIGH CURRENT SWITCHES AND CONDUCTORS
- HIGH VOLTAGE SWITCHES
- CAPACITORS



STATUS



THE GOALS

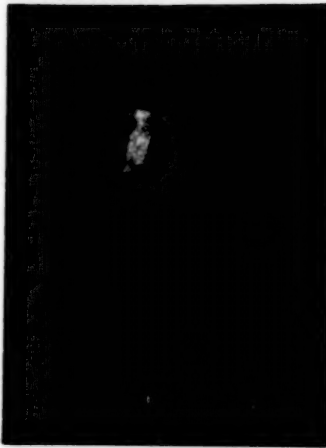
- HIGH CURRENT SWITCHES
- 2 X 10⁴ AMPERES
- 100 VOLTS
- HIGH VOLTAGE SWITCHES
- 150,000 VOLTS
- 500 - 1000 AMPERES
- CAPACITORS
- 500 VOLTS / POUND
- 10⁴ DISCHARGES

High current and high voltage switch technology requires significant advances to meet some of the expected high power future system needs. Increased capacitor energy density is also a major objective.

SWITCH TECHNOLOGY



SWITCH TESTING
APPARATUS



HIGH CURRENT
ROTARY SWITCH

This rotary switch has been demonstrated in-house at the Aero Propulsion Laboratory at 12 kilovolts and 60 hertz.

SUMMARY

- IMPROVED EFFICIENCY WITH MULTI-BANDGAP TECHNOLOGY
- HIGHER POWER AND SURVIVABILITY WITH LIGHTWEIGHT PLANAR AND CONCENTRATOR ARRAYS
- LARGE SIZE NICKEL-HYDROGEN CELL TECHNOLOGY
- 2.3X ENERGY DENSITY IMPROVEMENT WITH SODIUM SULFUR
- DUAL ROLE: TRANSPORT AND REJECTION
- SEVERAL ADVANCED CONCEPTS
- AMBITIOUS COMPONENT IMPROVEMENTS REQUIRED
- MISSION ENABLING

PHOTOVOLTAICS

ELECTROCHEMICAL POWER SOURCES

THERMAL MANAGEMENT

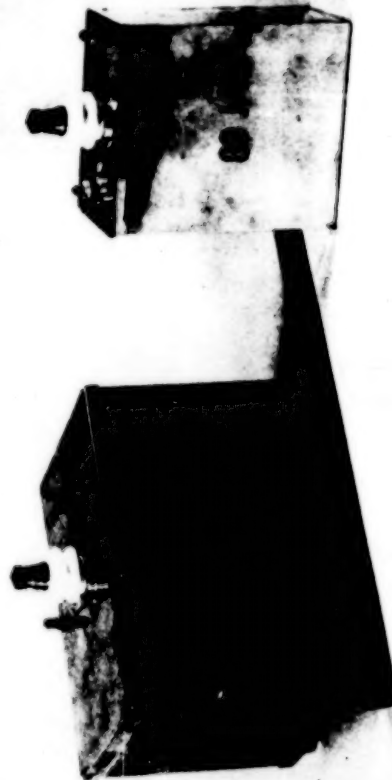
POWER CONDITIONING

In summary, the Air Force program is emphasizing technology which will permit higher power levels at reasonable weights and with enhanced survivability. Thermal management is likely to be a "show stopper" for very high base load and pulsed power systems. Advanced power processing technology requirements present some challenging objectives. Significant resources will be required to achieve those objectives.

GOALS	FY83	FY84	FY85	FY86	FY87	FY88
HIGH CURRENT SWITCHES AND CONDUCTORS	10 KA SWITCH	500 KA SWITCH	2 MA SWITCHING TECHNOLOGY			
HIGH VOLTAGE SWITCHES	E-BEAM SWITCH	40 KV 40 KA THYRISTRON (ARMY)	INSTANT START THYRISTRON (ARMY)	150 KV 500 KA SWITCH TECHNOLOGY		
CAPACITORS	80 JOULES POUND			500 JOULES POUND 10' LIFE		

Switch programs are directed toward high current capability approaching 2 mega-amperes. High voltage switch efforts approaching 150 kilovolts are also planned. Increased capacitor energy density and life will require improved dielectrics.

HIGH ENERGY DENSITY CAPACITORS



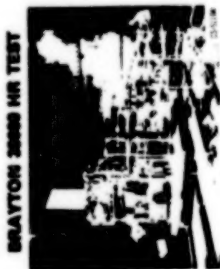
Earlier capacitor work resulted in a factor of 4 improvement in energy density.

SP-100 SPACE POWER PROGRAM

Judith H. Ambrus
NASA Headquarters

No text available at time of printing.

BACKGROUND



- SPACE NUCLEAR REACTORS HIGH LEVEL ACTIVITY FROM MID 50S TO EARLY 70S
- FLIGHT EXPERIENCE WITH NPS
 - ONE U.S. REACTOR (SNAP 10A, 1965, 800W)
 - >20 SOVIET ROSSAT MISSIONS
 - RTGs 12 NASA, 9 DOD, MISSIONS (NEXT GALILEO, ISPM)

- "KEEP ALIVE" PROGRAMS FROM 1973-1983
 - THERMAL-TO-ELECTRIC CONVERSION (NASA)
 - HEAT PIPE REACTORS (NASA/DOE)

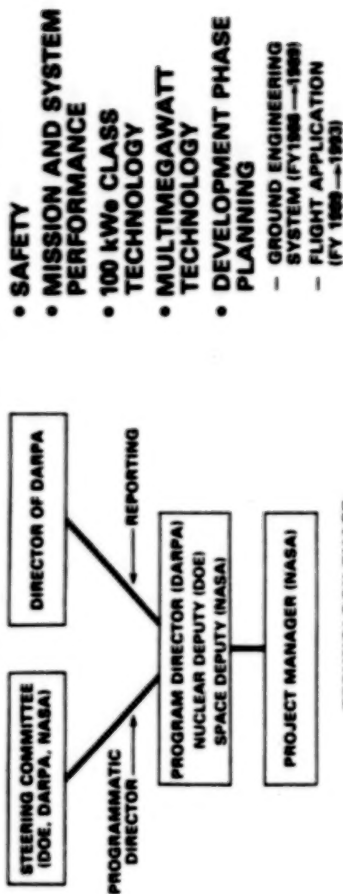
- TRI-AGENCY WORKING GROUP (FORMED 1980)
 - ASSESSED NEED FOR SPACE NUCLEAR REACTOR TECHNOLOGY PROGRAM

- MEMORANDUM OF AGREEMENT (FEB. 1983)
 - NASA/DARPA/DOE
 - ESTABLISHED SP-100 PROGRAM



Figure 1.

SP-100 PROGRAM



TECHNOLOGY PHASE

	FY 83	FY 84	FY 85
MEMORANDUM OF AGREEMENT INTERMEDIATE REVIEW GES PHASE DESIGN	▼	▼	▼
BUDGET (TOTAL M\$)	11.4	18.9	18.9
BUDGET (NASA, M\$ + DMY)	2.9 + 10	3.9 + 15	3.9 + 18

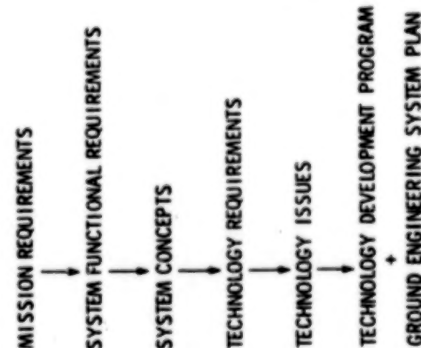
Figure 2.

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SP-100

Focused Technology Program

LOGIC FLOW



ITERATION

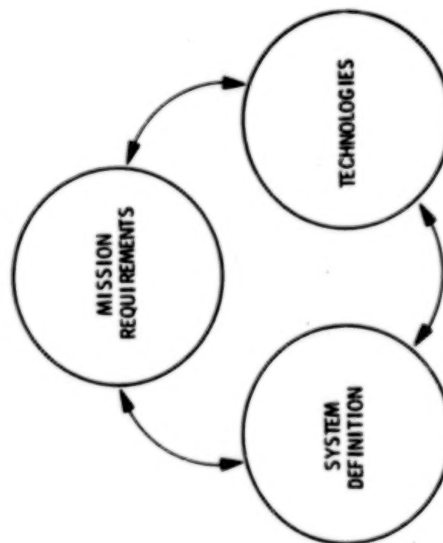


Figure 3.

SP-100

Technology Strategy

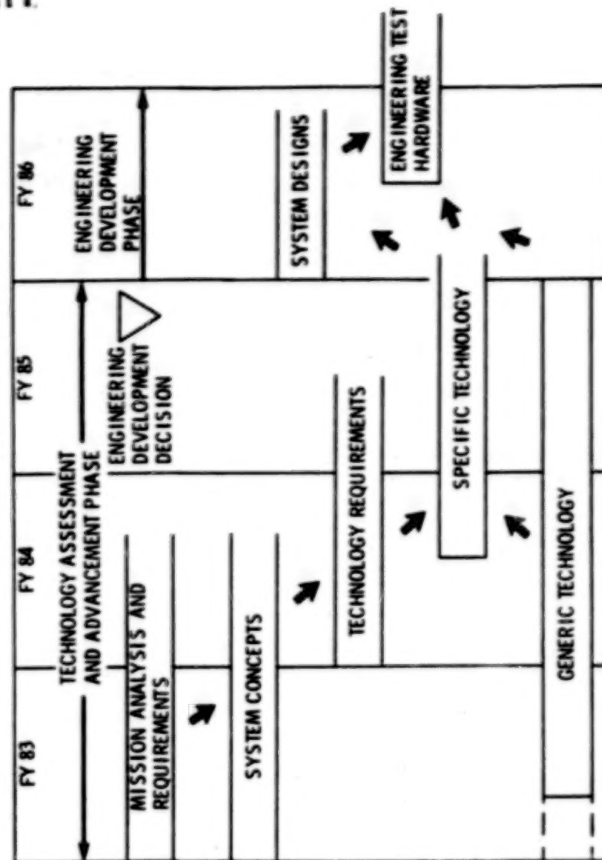


Figure 4.

CIVIL MISSIONS ENABLED OR ENHANCED BY NUCLEAR REACTOR POWER

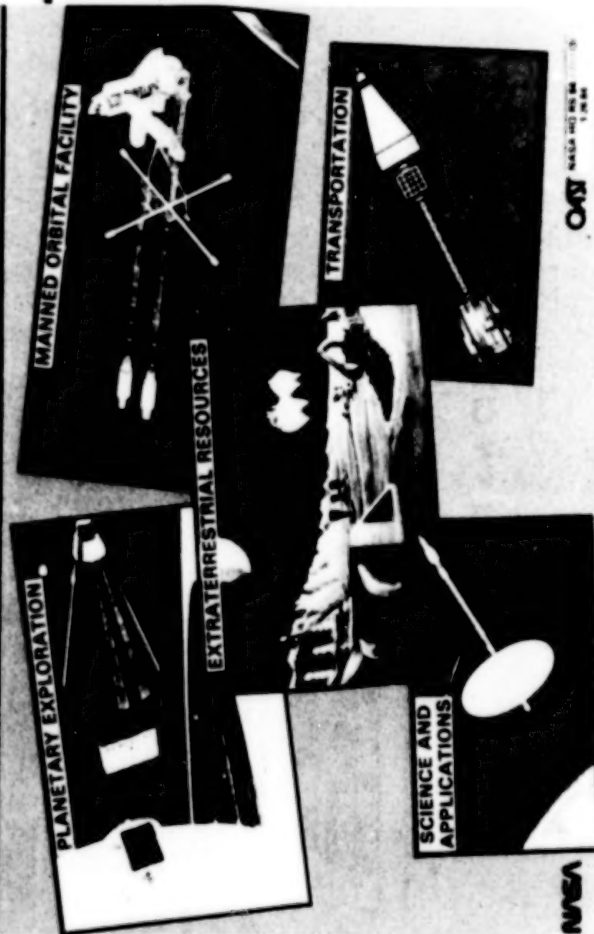


Figure 5.

100 KW CLASS SUMMARY OF SYSTEM FUNCTIONAL REQUIREMENTS

• ELECTRIC POWER OUTPUT	100 KW
• GROWTH	TO ~ 1 MW IN SHUTTLE
• MASS	3000 KG
• SIZE	1/2 SHUTTLE BAY (LEAVES ROOM FOR PAYLOAD, OTV)
• DESIGN LIFETIME	7 YEAR OPERATING 10 YEAR TOTAL
• EARLY DEMONSTRATED LIFE	2 YEARS FOR FIRST APPLICATIONS
• RADIATION TO USER	10^{13} N/CM ² (1 Mrad) 5×10^5 RADS

Figure 7.

MISSIONS REQUIRING CONTINUOUS 100 KW CLASS SPACE POWER

NEAR TERM (1990-2000)

- COMMUNICATIONS
 - COMMERCIAL (DIRECT BROADCAST)
 - MILITARY (HIGH DATA RATE, JAM FREE, MORE SURVIVABLE)
- RADAR SURVEILLANCE & REMOTE SENSING
 - MILITARY (LOW CROSS SECTION, HIGH SEARCH RATE, MORE SURVIVABLE)
 - CIVIL (AREA AIRCRAFT CONTROL, GLOBAL WEATHER FORECASTING)
- GROWTH SPACE STATION
 - MANUFACTURING
 - HOUSEKEEPING

FUTURE MISSIONS

- SPACE TUG
- OUTER PLANET MISSIONS (PROPULSION, SENSING)
- TRACKING SYSTEMS
- DESIGNATION SYSTEMS
- HOUSEKEEPING & CRYOGENICS
- SPACE EXPLOITATION (LUNAR BASE, ETC.)

Figure 6.

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100 KW, TECHNOLOGY SELECTIONS

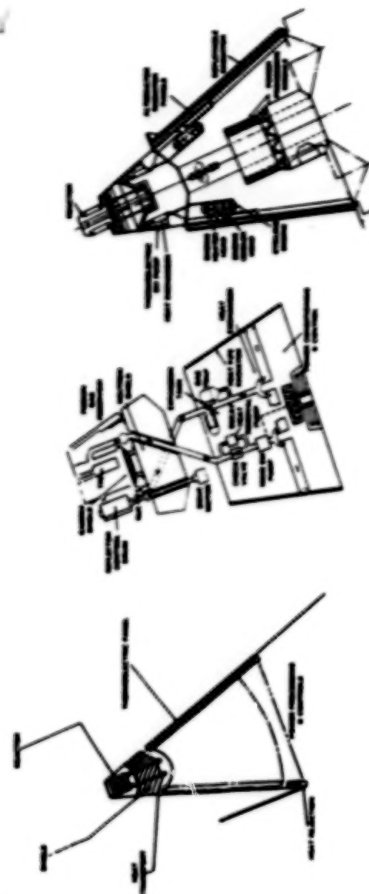


Figure 8.

NUCLEAR SPACE POWER TECHNICAL ISSUES

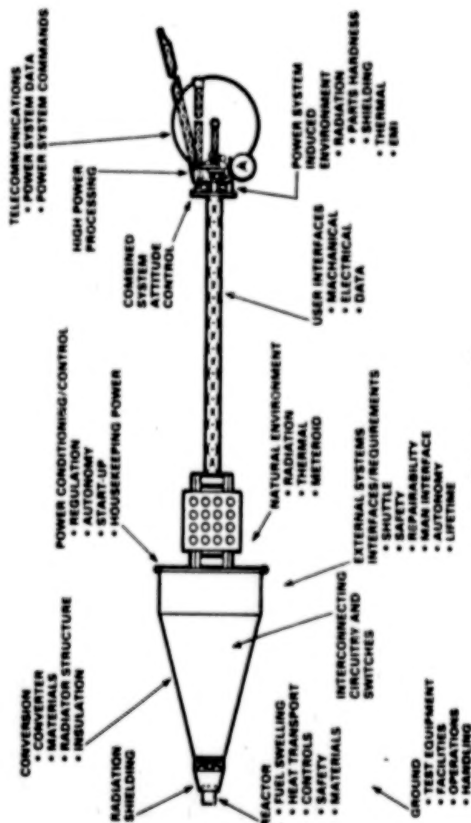


Figure 9.

100 KW CLASS SCHEDULE

84	85	86	87	88	89	90	91	92
----	----	----	----	----	----	----	----	----

TECHNOLOGY PHASE

SELECT CONCEPT
DEMONSTRATE TECHNICAL FEASIBILITY
DEMONSTRATE SAFETY FEASIBILITY

ENGINEERING DEVELOPMENT & GROUND TEST PHASE

PRELIMINARY FLIGHT SYSTEM DESIGN
ALL ENGINEERING DATA
SELECT MISSION & PAYLOAD

COMMITMENT TO
FLIGHT APPLICATION PHASE

FLIGHT APPLICATION PHASE

Figure 11.

MAJOR 100 KW CLASS TECHNOLOGY ISSUES & ACTIVITIES



Figure 10.

SDI OBJECTIVES

- LONG TERM R&D PROGRAM TO GUIDE FUTURE DECISIONS ON STRATEGIC DEFENSE
- EXAMINE FEASIBILITY OF A SYSTEM FOR BALLISTIC MISSILE DEFENSE
- EARLY DEMONSTRATION OF KEY TECHNOLOGIES NEEDED FOR EFFECTIVE DEFENSE

Figure 12.

SDI ORGANIZATION

- STRONG OSD CENTRAL MANAGEMENT—SINGLE PROGRAM MANAGER
- SERVICES AND AGENCIES PARTICIPATE IN BUDGET FORMULATION PROCESS AND WILL BE RESPONSIBLE FOR EXECUTING PROGRAMS
- FIVE MAJOR PROGRAM ELEMENTS:
 - SURVEILLANCE, ACQUISITION, TRACKING AND KILL ASSESSMENT
 - DIRECTED ENERGY WEAPONS
 - KINETIC ENERGY WEAPONS
 - SYSTEMS ANALYSIS AND BATTLE MANAGEMENT
 - SUPPORT PROGRAMS

Figure 13.

MMW TECHNOLOGY PROGRAM PLANNING PARTICIPANTS

- AIR FORCE
 - AIR FORCE WEAPONS LABORATORY
 - AIR FORCE AEROSPACE PROPULSION LABORATORY
 - AIR FORCE MISSILE PROPULSION LABORATORY
 - AIR FORCE SPACE TECHNOLOGY CENTER
- NASA
 - LEWIS RESEARCH CENTER
- DEPARTMENT OF ENERGY
 - DOE
 - OAK RIDGE NATIONAL LABORATORY
 - SANDIA NATIONAL LABORATORY
 - LOS ALAMOS NATIONAL LABORATORY
 - BROOK HAVEN NATIONAL LABORATORY
 - ARGONNE NATIONAL LABORATORY
 - LAWRENCE LIVERMORE NATIONAL LABORATORY

Figure 15.

STRATEGIC DEFENSE INITIATIVE SPACE PRIME POWER & POWER CONVERSION PROGRAM SCOPE

SP-100 ORGANIZATION FORMED FROM THREE AGENCIES TO ENSURE PROPER ATTENTION GIVEN TO:

- NUCLEAR & AEROSPACE SAFETY
- DOD MISSIONS
- NASA, CIVIL, & COMMERCIAL MISSIONS
- NUCLEAR TECHNOLOGY AND ENGINEERING
- POWER TECHNOLOGY AND ENGINEERING
- AEROSPACE TECHNOLOGY AND ENGINEERING

SCOPE OF CHARTER:

- 100 KW CLASS (10 TO 1000 KW) (NUCLEAR)
- MEGAWATT & MULTI-MEGAWATT CLASS (NUCLEAR & NON-NUCLEAR)

Figure 14.

SPACE POWER REGIMES

ELECTRIC POWER LEVEL	DURATION		
	CONTINUOUS	QUASI-CONTINUOUS	BURST
10-1000 KW ("100 KW CLASS")	COMMERCIAL NASA/CIVIL MILITARY (SDI & NON-SDI)		
LOW MW'S	MILITARY (SDI) CIVIL (DISTANT FUTURE)	MILITARY (SDI)	
MMW			MILITARY (SDI WEAPONS)

Figure 16.

STRATEGIC DEFENSE INITIATIVE MULTI-MEGAWATT SPACE POWER TECHNOLOGY PROGRAM PLAN

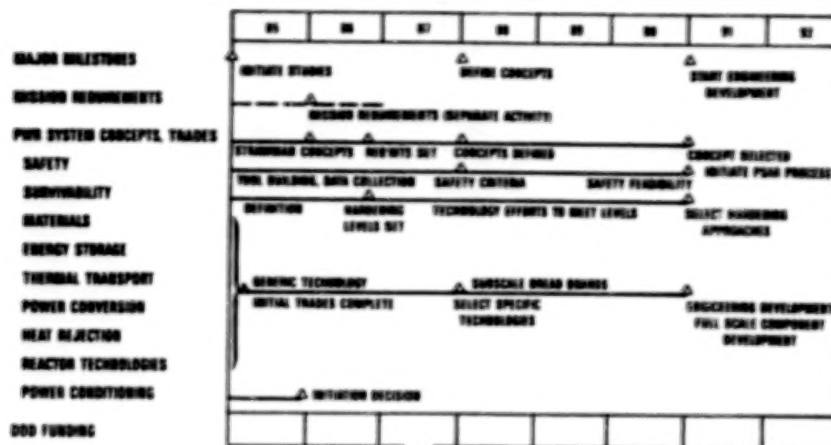


Figure 17.

RADIOISOTOPE SPACE POWER PROGRAMS

Bernard J. Rock
U.S. Department of Energy

Radioisotope Thermoelectric Generators (RTGs) have been extensively used in past space missions with great success. An improved generation of RTGs employing a new light weight, modular heat source is being built by the DOE for NASA launches in 1986. More advanced modular RTGs which promise power-to-weight ratios of over 8.8 watts(e)/kg (4 watts(e)/lb) are currently under development by the DOE and could be flight-ready within five years.

Dynamic Isotope Power Systems (DIPS) which offer 18-20% conversion efficiencies have been demonstrated by the DOE in ground tests. DIPS would be useful for several military missions requiring power in the low-kilowatt range. These systems could also be brought to flight readiness by the DOE following receipt of firm user requirements.

INTRODUCTION

The Office of Special Nuclear Projects (OSNP) at the U.S. Department of Energy (DOE) Headquarters in Germantown, MD is responsible for all of the DOE's space nuclear power programs involving the use of radioisotope energy sources. As the Director of that Office, I am pleased to present to the participants of this Space Power Workshop what is currently being done and what is planned for the foreseeable future in the DOE's radioisotope space power programs.

Radioisotope space power systems, in particular the Radioisotope Thermoelectric Generator (RTG), are an established technology to be considered by space mission planners and spacecraft designers. 34 RTG power systems have been successfully used in space over the past two decades. They have demonstrated characteristics of long life, high reliability, adaptability to extreme mission environments, and safe handling and use. RTGs have been successfully launched on everything from a Scout to the Saturn V launch vehicle; used in missions in earth orbit, to the Moon, to the outer planets and beyond; and, integrated into manned and unmanned space applications. This record not only confirms the versatility of the RTG, but also the ability of the DOE (and its predecessors) to support various user agency requirements.

The use of RTGs has been directed to support experimental military and civilian earth orbiting satellites or to NASA missions to the outer planets or on the surfaces of the Moon and Mars - where they were clearly the best system available. One of the most significant advantages of radioisotope space power systems, their military utility, has yet to be fully exploited. It is anticipated that radioisotope power systems will be needed to enable or enhance future military operations in space, as

well to continue to provide power for scientific planetary missions, etc. It is because of these anticipated needs the DOE is continuing to support its radioisotope space power systems development programs.

MISSION REQUIREMENTS

Evolving mission requirements show that the spacecraft of the future will need higher power levels than those of the past. These trends are shown in Figure 1. In addition, the U.S. space program is maturing to where payloads are not only expected to do more, but to do it reliably over longer lifetimes at improved cost effectiveness.

Military Space System Technology Plans clearly indicate needs for higher power levels, more survivable systems, more flexibility in operational orbits (altitude and inclination), and increased capability in on-orbit maneuvering and spacecraft pointing.

The DOE has ongoing programs in three areas to meet these projected needs with radioisotope space power systems. The three major DOE space programs are:

- (1) The Galileo and ISPM flight generator program,
- (2) The Modular-RTG ground demonstration program, and
- (3) The Dynamic Isotope Power System (DIPS) development planning effort.

The first program involves the current generation of RTGs to be used on two up-coming NASA space science missions and the last two programs directly address the higher power requirements projected for future civilian and military missions. Each will be discussed further.

GENERAL PURPOSE HEAT SOURCE

All currently planned radioisotope space power systems will employ the General Purpose Heat Source (GPHS), which has been under development by the DOE for several years. The GPHS is a modular device, as illustrated in Figure 2. Each module is designed to protect its Plutonium-238 fuel encapsulation and to minimize fuel dispersal under postulated mission abort scenarios. Each module measures approximately 9.7 x 9.3 x 5.3 cm (3.8 x 3.7 x 2.1 in) and weighs 1.45 kg (3.2 lb). It contains 250 thermal watts of Plutonium-238 oxide fuel in four 62.5-watt pellets separately encapsulated in thin-walled Iridium alloy containment shells. Two fuel capsules are encapsulated in a graphite impact shell and two of these impact members are surrounded by graphite insulators within an aeroshell so as to protect the fuel capsules against high heat pulses, e.g. during postulated aborts leading to reentry into the earth's atmosphere, and to cause the fuel capsule to impact at tolerable temperatures and velocities.

The selection of 250 thermal watts per module was made to provide a high degree of modularity. This assists the system designer in matching the thermal power of the total heat source to the mission's electrical power requirements by assembling the proper number of GPHS modules. The GPHS also has a much higher power-to-weight ratio

than previous heat source designs.

The GPHS has that name because it is applicable to any one of a number of power conversion systems or uses. Thus, it can be used in RTGs; it can be used in Dynamic Isotope Power Systems (DIPS); and, it can also be used where only heat is required, e.g. powering a thermally activated cryogenic cooler.

The GPHS module will be the heat source of choice for all radioisotope powered space missions for at least the next decade and perhaps longer. After obtaining basic safety approval for use on the shuttle-launched NASA missions in 1986, obtaining safety approvals for its use on future shuttle missions should be routine. The GPHS is a prime example of where standardization, modularization and repetitive use of a proven design could save future development costs as well as the recurring costs associated with safety approvals.

CURRENT RTG FLIGHT PROGRAMS

The current generation of space RTGs under development by the DOE is called the GPHS-RTG, because it is a thermoelectric generator built around the new GPHS. (See Figure 3.) The GPHS-RTG program is currently in the qualification testing phase.

The flight units now being built and fueled are to be launched by NASA in 1986. Two units will be aboard the Galileo spacecraft to explore the planet Jupiter and its environs. Another unit will be aboard the International Solar Polar Mission (ISPM) spacecraft, which is being built by the European Space Agency (ESA) and is to investigate the sun from high solar latitudes. Both the NASA-Galileo and the NASA/ESA-ISPM programs are committed to the successful use of the GPHS-RTG.

These missions will be the first launches of RTGs by the space shuttle. The existing shuttle cooling system will protect the generator, the payload(s) and the shuttle by removing the heat generated by the RTG(s) while in the payload bay. All shuttle interfaces and safety related issues will be demonstrated prior to the 1986 launch date.

The goals of the GPHS-RTG program are to provide a shuttle-compatible RTG with a specific power of 5.2 watts/kg (2.4 watts/lb) and a 7 year life.

The GPHS-RTG is built around a central heat source consisting of a stack of 18 GPHS modules producing a nominal output of 4500 thermal watts. The heat source is radiatively coupled to the hot shoes of the surrounding silicon-germanium thermoelectric elements which are attached to a finned aluminum generator housing. The thermoelectrics are the same materials successfully used on previous long-lived space RTG missions. Operating between a hot junction temperature of 1273 K (1000 C) and a cold junction of 573 K (300 C), each GPHS-RTG will produce about 293 electrical watts (at 28-30 volts) for a system efficiency of about 6.5%. Each RTG will weigh 55.9 kg (123 lbs). The GPHS-RTG envelope size is 113.0 cm (44.5 in) long by 42.2 cm (16.6 in) O.D., including fins.

ADVANCED RTG PROGRAM

The DOE also has an advanced RTG under development called the Modular-RTG. Its two main features are its modularity, in both heat source and thermoelectric converter, and its much higher specific power. The General Electric Company has a three year contract to provide to DOE a Ground Demonstration System (GDS) for a Modular-RTG. This work is an outgrowth of the Modular Isotope Thermoelectric Generator (MITG) design effort conducted by Fairchild Space Company and reported in the IECEC Proceedings for 1980, 1981, and 1983. The primary goal of this program is to develop the most efficient packaging for the GPHS module and, consequently, the highest possible generator specific power for a device using this heat source. A second goal is to provide a high degree of modularity so that variations in mission power requirements can be accommodated by a minimum of RTG redesign and development effort.

The modularity of this type of generator is illustrated in Figure 4. The RTG is made up of identical modular slices of generator around one GPHS module. Each slice is expected to produce about 20.5 watts (electrical) at the full 28 volt system level. Once the basic generator module has been developed and qualified, a new RTG can be designed by stacking up the modules and adding the end sections.

The Modular-RTG will include more thermoelements than previous RTGs to meet the voltage output with redundancy for reliability. It will also take advantage of a gallium phosphide additive to lower the thermal conductivity of the silicon germanium thermoelectric material and thus increase its figure of merit by almost 10%.

Prototypical modules of this design have been built and are undergoing performance tests. Cold vibration tests have been successfully completed on thermoelectric components and hot vibration tests are being prepared.

Design analyses of a typical 14 module RTG project a generator weight of only 30.9 kg (68 lb) for a power output of 288 electrical watts. This corresponds to a system efficiency of 8.2% and a power-to-weight ratio of 9.3 watts/kg (4.2 watts/lb). This improvement in specific power (78% higher than for the GPHS-RTG) is a major motivation for the development of these advanced RTGs.

The modular RTG design can be scaled up to about 500 watts (electrical), corresponding to 24 GPHS and generator modules. Longer stacks of heat sources would require too much axial preload to withstand vibration loads during launch. Therefore, for higher power levels it probably makes more sense to use multiple RTGs. This could also aid in integrating them with the spacecraft design.

Although there is no inherent power limit in the use of RTGs aboard a single spacecraft, for radioisotope-powered spacecraft requiring much more than a kilowatt, the use of dynamic conversion systems becomes attractive because their higher conversion efficiencies permit better use of the radioisotope fuel.

DYNAMIC ISOTOPE POWER SYSTEMS

Review of the Military Space System Technology Plan (MSSTP) shows a USAF need for power in the range from 1 to 10 kilowatts (electric). Missions 2, 3, 6, 13 and 22 of the MSSTP are especially suitable for Dynamic Isotope Power Systems (DIPS) utilization.

A number of different dynamic conversion systems, including Brayton, Rankine, and Stirling, have been investigated in previous DOE space power programs. Space-configured radioisotope systems have been designed, built, and tested for all three types of conversion cycles.

The feasibility of the isotope Brayton and Rankine systems was confirmed by ground demonstration tests that were completed in 1978. Garrett/AiResearch acted as the system contractor for the Brayton System, and Sundstrand as the system contractor for the Organic Rankine System.

The Brayton System was built of superalloys, and employed a helium-xenon inert gas mixture as its working fluid. It operated at a turbine inlet temperature of 1025 K (752 C) and a compressor inlet temperature of 328 K (55 C). At these temperatures, it has a system conversion efficiency of 20%.

The Organic-Rankine System, shown in Figure 5, uses Dowtherm as its working fluid, with a turbine inlet temperature of 644 K (371 C) and a pump inlet temperature of 373 K (100 C). The turbine inlet temperature is limited by the decomposition of the organic working fluid at higher temperatures. The system demonstrated a conversion efficiency of 15%, which can be raised to 17% by improvements that have since been identified.

In both of these dynamic systems, there is only a single moving part. This is a solid member which forms the combined rotating unit. In the case of the Brayton system, it consists of a turbine wheel, an alternator rotor, and a compressor wheel on a single shaft. This rotating unit rides on a gas film via foil bearings, and does not contact any stationary material. Thus, there should be no rubbing or wear mechanism. In the Organic Rankine system, the turbine-alternator-pump unit is lubricated and cooled by its own organic working fluid for reliable operation.

In 1978 the DOE selected the Organic Rankine approach over the Brayton approach to provide a DIPS for a planned Air Force-sponsored space flight test, which never materialized. The DIPS program was continued for two more years through a technology verification test phase which was last funded in January 1981. Figure 6 shows the Organic Rankine DIPS at the thermal vacuum test facility. The effort was terminated due to the lack of an identified mission use.

The DOE has recently been engaged in a planning exercise to re-establish the DIPS technology effort. This activity is in support of DOD/USAF requirements. A detailed DIPS program plan has been developed and presented to USAF personnel. The summary schedule for the DIPS development program is shown in Figure 7. There appear to be no technology barriers to the development and deployment of a DIPS system.

CONCLUDING POINTS

The key points I have tried to make in this presentation are:

- o Many static radioisotope power systems have been successfully used in space, with high reliability.
- o Advanced static systems with higher powers and lighter weights are under development.
- o Flight-configured dynamic isotope power systems have been demonstrated in ground tests.
- o Nuclear power can enable survivability of military space systems, because: They can be much harder, and they can make the rest of the system more survivable.
- o Advanced static and dynamic systems can be flight-ready by the late 1980's/early 1990's.
- o They will be shuttle-compatible.

Some additional points, which I have not covered in detail, but which should be considered by potential users of radioisotope space power systems are:

- o Their use does not conflict with any US or UN policies or treaties.
- o There are no policies restricting allowable orbits.
- o Adequate safety is demonstrable.
- o There is a well-established safety approval process.

SPACE NUCLEAR POWER & MISSION REQUIREMENTS

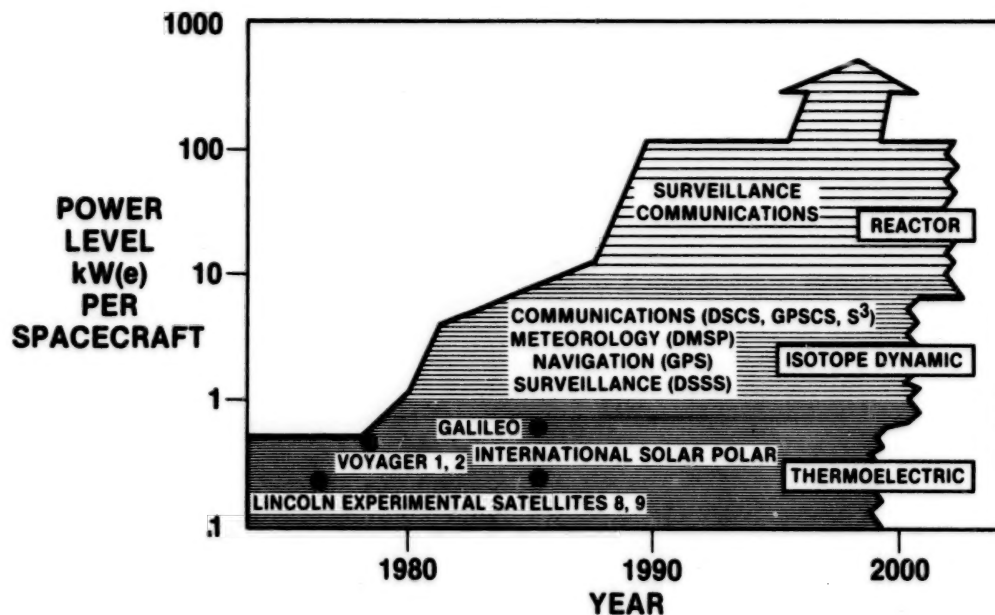
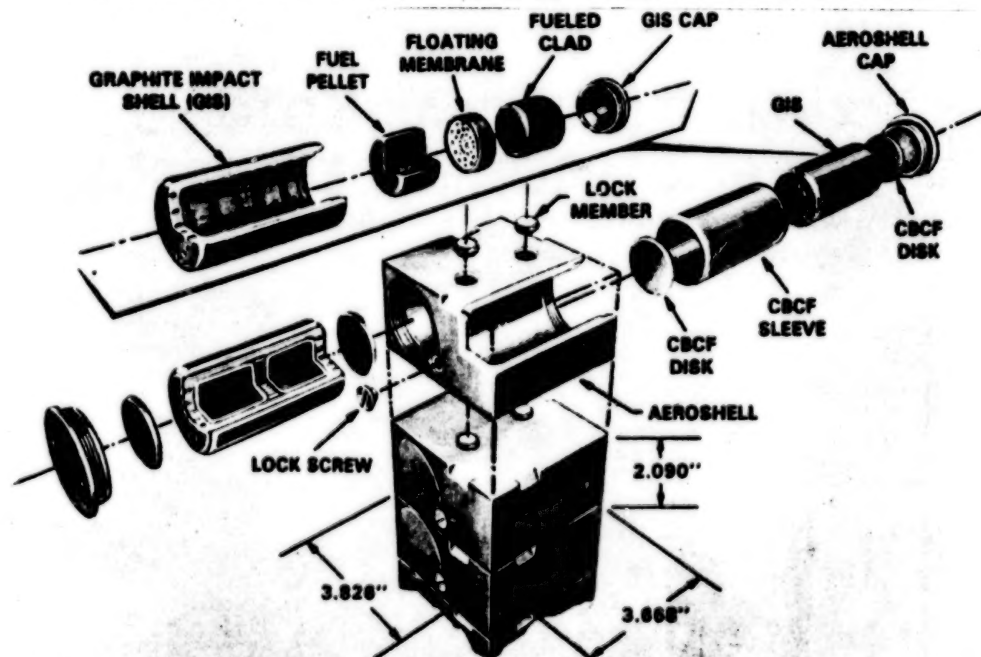


FIGURE 1

GENERAL PURPOSE HEAT SOURCE

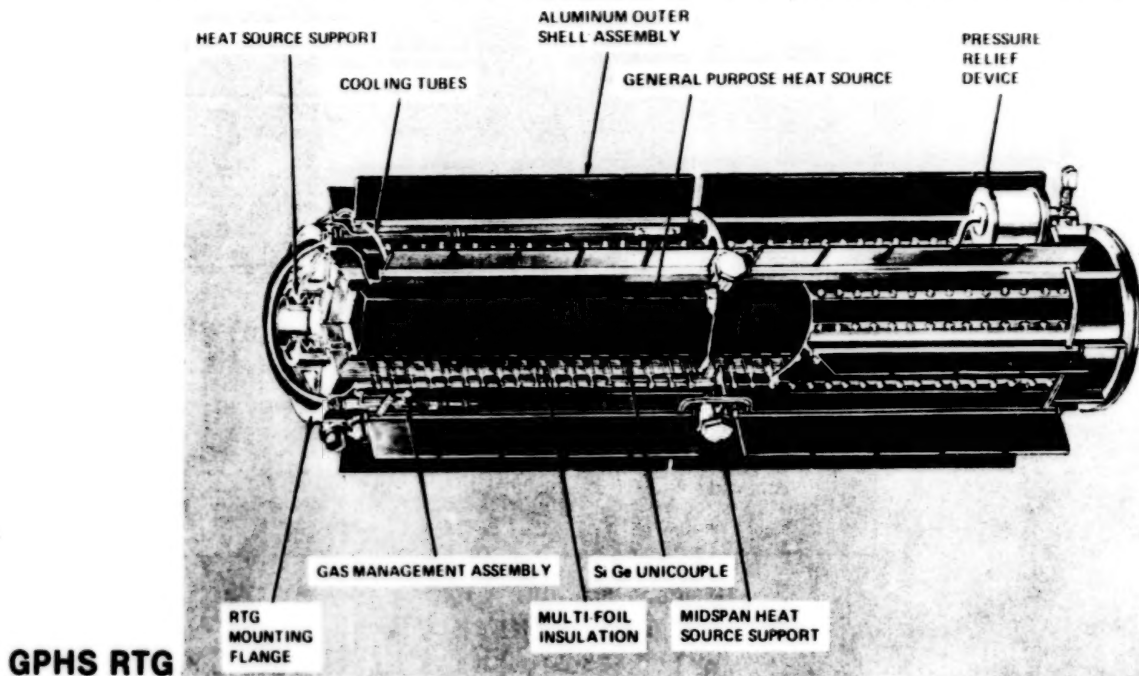


- MODULAR-MEETS BROAD RANGE OF POWER REQUIREMENTS
- STATIC AND DYNAMIC SYSTEMS APPLICATIONS
- INCREASED POWER PER POUND

FIGURE 2

ORIGINAL PAGE IS
OF POOR QUALITY

THERMOELECTRIC SYSTEMS POWER FOR CIVILIAN & MILITARY NEEDS



GPHS RTG

- 285 WATTS(e), 123 LBS
- COMPATIBLE WITH SHUTTLE
- NEW HEAT SOURCE FOR INCREASED POWER PER POUND
- IMPROVED SAFETY

FIGURE 3

MITG (MODULAR ISOTOPIC THERMOELECTRIC GENERATOR)

ILLUSTRATIVE GENERATOR

14 Slices, 288 Watts, 68 Lbs., 4.25 W/Lb

MODULAR SLICE

(20.5 Watts at 28 Volts)

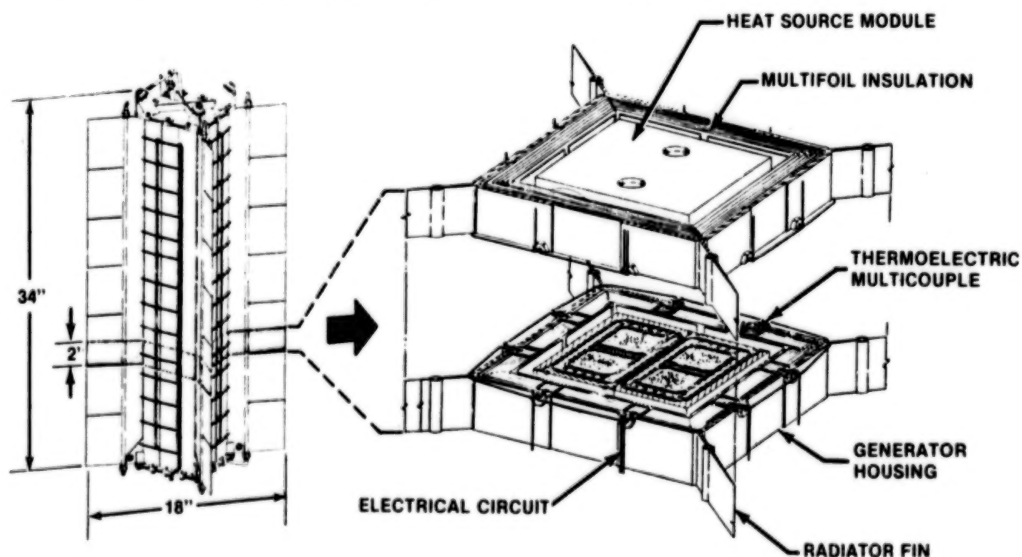


FIGURE 4

POWER FOR MILITARY DYNAMIC ISOTOPE POWER SYSTEM

- UP TO 2 kW(e) PER MODULE
- MULTIPLE MODULE APPLICATIONS
- TECHNOLOGY READY
- EARLIEST FLIGHT 5 YEARS AFTER START
- MULTIMISSION APPLICABILITY

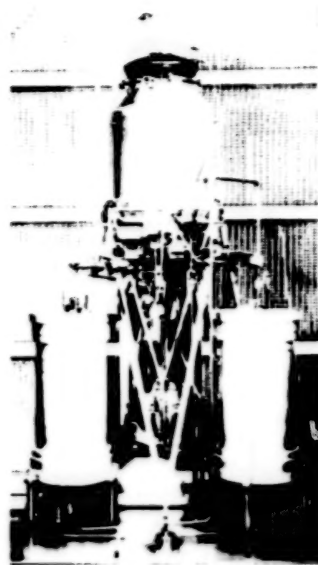


FIGURE 5

DIPS VERIFICATION TEST UNIT

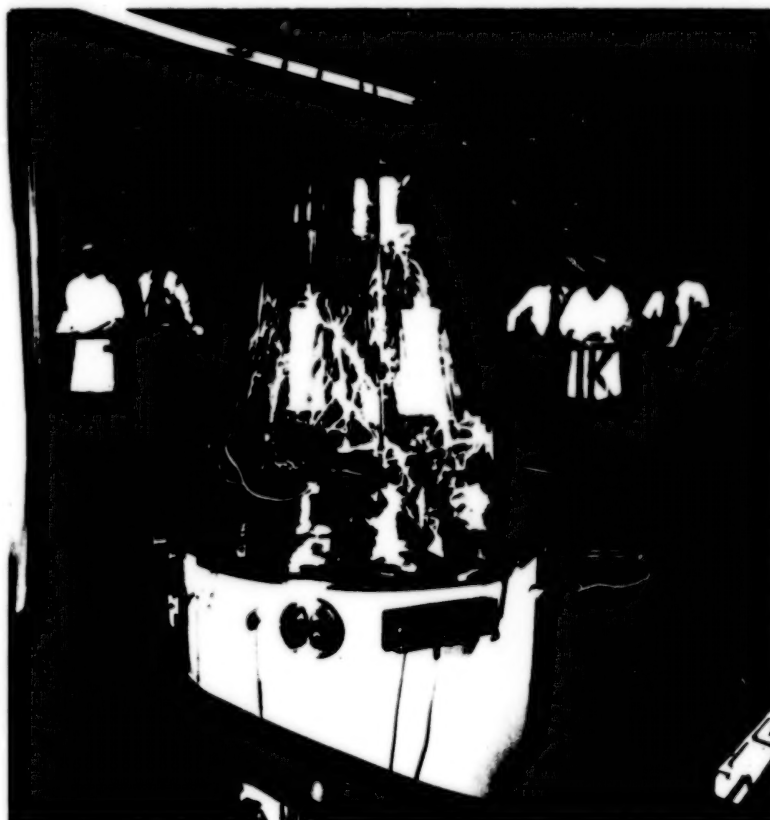


FIGURE 6

DYNAMIC ISOTOPE POWER SUBSYSTEM (DIPS) SUMMARY SCHEDULE FOR POWER SUBSYSTEM

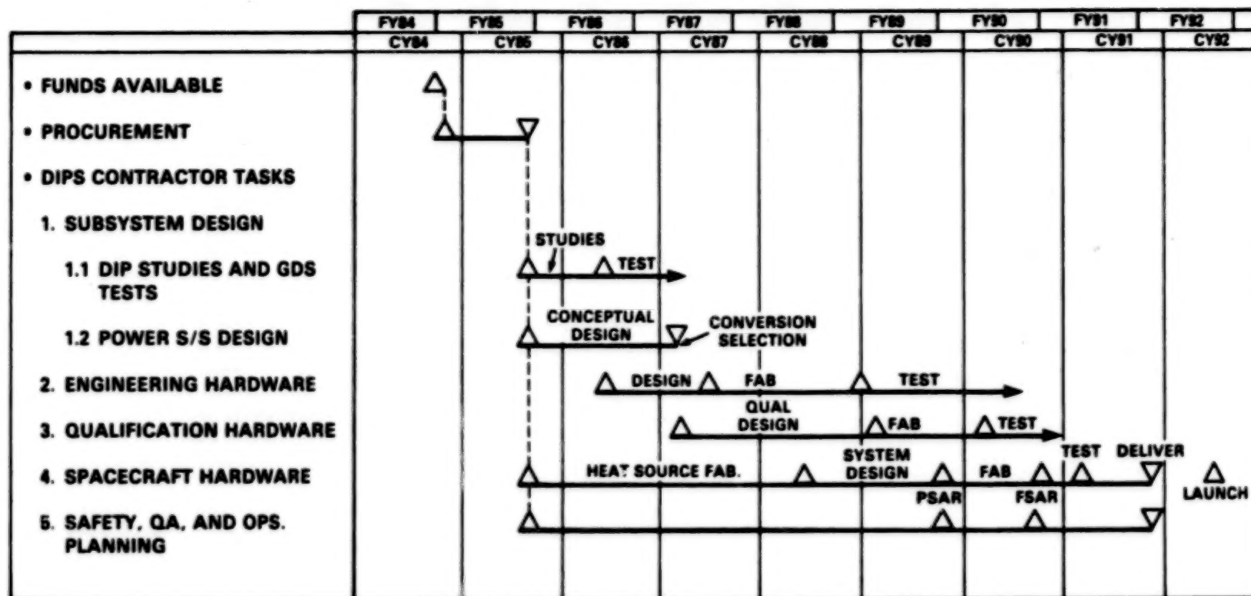


FIGURE 7

TRENDS IN SPACE PHOTOVOLTAIC TECHNOLOGY*

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The current status of silicon and gallium arsenide (GaAs) solar cell technology is described, and anticipated near and far term projections of photovoltaic cell performance are provided. It is shown that current ultrathin silicon and near term GaAs solar cells provide substantial enhancement of planar solar array performance. The advantages of utilizing GaAs cells in high concentration arrays is discussed. Evidence is provided to support the view that photovoltaics offers a viable means of supporting long term space objectives.

BACKGROUND

Solar photovoltaics is still a relatively young technology, now only in its third decade of use for space power applications. In the 1960s the silicon solar cell became the exclusive means of generating photovoltaic space power. This occurred because silicon had become the basic medium for the highly profitable semiconductor device industry. By drawing on the broad technology base that developed to support the growth of first the transistor, then the microelectronics industry, the silicon solar cell was able to compile an impressive record for reliable operation in space. Alternate materials such as CdS and GaAs could not compete since no equivalent technology base was available to support them.

During the 1970s silicon solar cell technology improved dramatically with respect to conversion efficiency, radiation resistance and configuration (size and thickness). This progress, to a large extent, was the result of an increase of technical personnel working directly in the silicon space cell area. Other factors such as additional technology transfer from the microelectronics industry and the strong motivation of users to retain the space proven silicon cell also were important.

* Portions of the research described in this paper presents the results of one phase of research carried out by the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautics and Space Administration.

The 70s also saw the rapid growth of the opto-electronics industry, which was based on compound semiconductors, and a national commitment to solar power, including photovoltaics. The first development led to the current GaAs solar cell, a direct outgrowth of light emitting diode technology. The second caused a shift in resources and personnel away from the silicon space solar cell field. Since the initial emphasis of terrestrial photovoltaic research and technology was cost reduction at the sacrifice of conversion efficiency and meeting a totally different set of environmental requirements, the momentum created by the technology surge of the early and mid 70s in silicon space cell progress was lost.

At present the primacy of the silicon solar cell for space applications is being challenged by GaAs which has already demonstrated superior conversion efficiency (16-18 vs 14-15 percent), greater resistance to most of the components of the natural space radiation environment and less sensitivity to the power degrading effects of elevated operating temperature. These advantages translate to an extremely significant margin in power output at end of mission life when compared to silicon. For this reason there is a great deal of interest on the part of some in the user community to bring GaAs to flight readiness. In parallel, there has been a shift in focus of the terrestrial photovoltaic program to efficiency. However, crystalline silicon is only one of a wide variety of options now being pursued.

Thus the issue of the 1980s in space photovoltaics will be the trade-off between the reliability and cost advantages of silicon and the performance benefits offered by GaAs. This will be done at the user level and the outcome of these trades will be determined by mission unique requirements. The prospects for the acceptance of GaAs solar cells for space will increase, regardless of the results of the initial trades, if the current emphasis on its technology continues.

In the following section of this paper, the current status of silicon, GaAs and the more exotic solar cell options will be discussed. Projections of progress, in each case, will be provided based on specific scenarios. The impact of the current and project status of photovoltaics on space array development will also be addressed.

STATUS AND PROJECTIONS

Silicon Solar Cells

Currently the average silicon space cell is approximately 10 cm² in area (equal use of 2x4 and 2x6 sizes), 250μm thick, with an initial 28°C conversion efficiency between 12 and 15 percent, depending on the particular mission application. In contrast, at the end of the 1960s, the 2x2 cm was on the verge of universal acceptance. It was 300 to 350μm thick and delivered 10 to 11 percent conversion efficiency at 28°C. This illustrates most of the major trends in silicon.

Near term mission applications will probably employ even larger cells (5.9 x 5.9cm MILSTAR cell) as well as smaller sizes (4-12 cm²) as thin as 50-75μm. There is no projected near term improvement from current levels of conversion efficiency. The wraparound contact configuration will likely achieve acceptance for space use, but not because of the original arguments (reduced panel assembling cost and improved efficiency) made for its development.

Although the ratio of power at end of life and initial conditions (P/P_0) has not improved as much as other cell characteristics for many specific mission radiation requirements, the efficiency improvements incorporated into silicon cells have not, on balance, caused a deterioration in P/P_0 . Most of the limited current research in the area of silicon solar cells is oriented to approaches that have the potential to improve cell radiation resistance. It is known that oxygen plays a major role in creating the radiation induced defects that degrade cell power output. The recent work done by the NASA Lewis Research Center in which counter doping of silicon with lithium was used to prevent the formation of oxygen associated defects is a good example of the type of research being supported.

Existing activities oriented toward improving rigid and flexible solar array performance will likely assure that such end of life enhancing cell technologies as low absorptance and ultrathin planar configurations are incorporated into space power systems. Unfortunately the acceptance of GaAs cells for space flight use will probably terminate those efforts aimed at improving the performance of silicon solar cells at end of life. Therefore further progress in developing refined ultrathin (<50μm) cell structures, optimized approaches to lowering cell operating temperature in space, innovative configurations like the vertical junction, and assessments of the effect of lithium on cell radiation behavior will probably cease.

As stated previously, the present advantage of silicon resides in its space heritage and lower cost compared to GaAs. It would be quite inappropriate to discuss the status and projections for silicon without addressing cost. Both NASA and DOD consider cost reduction to be a major driver for the success of future, more ambitious programs. The evidence provided by the terrestrial photovoltaic program argues strongly that there are only two factors that have a significant impact on cost; market demand and a degree of standardization.

The current annual manufacturing capacity of space qualified U.S. solar cell suppliers is conservatively estimated to be of the order of 250 to 500 kW. The implementation of automated processing is responsible for this dramatic increase in production capacity. Three factors created this situation; the development of solar cell compatible manufacturing equipment by the microelectronics and terrestrial solar cell industries, rising labor costs and more sophisticated cell processing operations. The reasons that this transition to automated cell production has not resulted in lower cell cost are: (1) the cost of capitalization has not been fully amortized, (2) under utilization of capacity, leading to higher overhead rates, and (3) non-standardized customer requirements which reduces the cost effectiveness of automation.

The decision to establish a permanently manned Space Station will no doubt have some impact on silicon solar cell technology development, especially if the station becomes operational within a decade. No firm power requirements exist, but present estimates range from 10 to 75 kW. Although a precise definition of orbit remains to be determined, there is little doubt that the station will function in an altitude region that places severe demands on solar array-battery power systems because of the eclipse periods the system will experience. Thus it is possible that there would be the need for a solar photovoltaic array designed to deliver two to three times the amount of power actually required by the Station.

Orbital drag consideration argue for a compact Space Station power system. This requires that the array use the highest efficiency cells possible. The amount of power that may have to be generated by the array demands a significant reduction in cost. The power system operating lifetime goal (10 years) and the probable orbit means that the array could experience up to 60 thousand thermal cycles in a worst case situation. This suggests that weldable cells might be necessary. It should be mentioned that a NASA Lewis Research Center sponsored program to assess the current industry capability for welding silicon solar cells was begun over a year ago. The STS lift capability and the non-stressing radiation environment implies no need for thinner or more radiation tolerant cells. Therefore the current thrust in silicon cell development will not be greatly enhanced by the Space Station program.

Gallium Arsenide Solar Cells

In the mid 1970s liquid phase epitaxy (LPE) was used to produce high efficiency GaAs solar cells. This heteroface structure employed an AlGaAs window layer grown over the GaAs cell junction in order to reduce the effect of surface recombination velocity. Progress in improving cell efficiency and radiation resistance was rapid once the influence of window layer thickness and junction depth were understood. Early prototype cells were flown as part of the NTS-2 experiment with encouraging results.

These successes focused a great deal of research and development resources into this technology area. Competing approaches for developing GaAs cells were quick to develop, driven by concerns about process control and manufacturing throughput. Organo-metallic chemical vapor deposition (OM-CVD) techniques were used to produce homo and hetero-junction GaAs cells which displayed similar potential for high conversion efficiency and greater tolerance to electron radiation. The main manufacturing issue now involves determining the best process (LPE or OM-CVD) for producing large numbers (>1000 equivalent 2x2 cm devices per week) of uniform cells in an economical manner. The results of the AFWAL sponsored MANTECH program will play a major role in this determination due to its cost, efficiency and production goals.

Current GaAs cells have efficiencies between 16 and 18 percent, can be made in 2x4cm size and are 300 to 350 μ m thick. This last characteristic is cause for concern since GaAs is approximately 2.3 times as dense as silicon and presently costs between 2 to 5 \$ per cm², depending on the quantity and quality (surface finish, etch pit density, etc.) required. A number of approaches aimed at eliminating or substantially reducing the amount of GaAs substrate material, upon which the solar cell structure is formed, are in progress.

Paradoxically GaAs cells can be extremely thin (approximately 5 to 10 μ m) and produce full conversion efficiency. Thus there is tremendous incentive to solve the GaAs substrate challenge. The CLEFT approach, which is a peeled film technique, has demonstrated greater than 15 percent conversion efficiency for cells as thin as 5 μ m. This method allows the substrate to be reused, thus significantly reducing the cost as well as producing a cell with an extremely high specific power (W/kg.). Another technique utilizes a silicon substrate upon which a single crystal germanium layer is grown by means of OM-CVD. Since there is very little lattice mismatch between germanium and GaAs, it should be possible to produce GaAs cells whose weight is determined by the thickness of the silicon substrate.

Figure 1 illustrates the results of an analysis that examined the impact of cell yield, GaAs substrate cost and end of life advantage on panel assembly cost. It was assumed that GaAs cell processing cost was equivalent to silicon, fifty percent of the GaAs substrates could be recovered from rejected cells and that panel assembly cost, excluding the cell, was approximately \$4.5/cm² (cover, adhesive, interconnect, panel substrate, labor, inspection, test, yield, etc.). The conclusion is that, even if the fifty percent yield goal of the MANTECH program can be achieved, it will be necessary that the cost of GaAs substrates be reduced to well below \$1/cm² in order to realize any cost advantage at the panel assembly level. This calculation cannot take into consideration the potential systems level benefits that might accrue to a satellite through the use of smaller solar panels.

Figure 2 shows the effect of various levels of end of life advantage on the weight of a rigid substrate solar panel. This analysis used data for the FLTSATCOM array which employed state of the art silicon solar cells. It was assumed that the substitution of GaAs for silicon cells would not change any other materials used in this panel. It is quite apparent that significant panel weight advantages can only be achieved by a substantial reduction in the thickness of current GaAs cells, regardless of the projected end of life power advantage of this cell option. The situation with respect to flexible substrate panels will more than likely require even thinner cells or higher end of life advantages in order to justify the use of GaAs solar cells.

The use of end of life power advantage as a critical parameter to justify GaAs solar cells logically leads to one of the major issues associated with this option. Although 1 MeV electron radiation tests clearly demonstrate the superiority of GaAs for fluence levels likely to be encountered by most missions, the question of the influence of the space proton radiation environment has not been completely resolved. In the case of the silicon cell, which is a bulk dominated device, the equivalence between the effects of

1 MeV electrons and protons of various energies has been established. This information is not yet available for the GaAs cell. Since only the first 5 to 10 μm of this device are essential for power output, extrapolation of silicon derived data cannot be used to predict the radiation behavior of GaAs cells. It will be necessary, once the rapidly evolving GaAs cell technology stabilizes, to generate data on the relationship between proton and electron effects in order to allow panel design tradeoffs to be made between GaAs and silicon.

The use of concentration offers an across the board solution to all the perceived problems associated with GaAs. The higher initial conversion efficiency and reduced susceptibility to the effects of elevated temperature operation offered by GaAs enables the utilization of relatively high (>20X effective) concentration systems. These systems essentially eliminate the impact of individual cell cost and weight on panel performance. The inherent shielding associated with producing high solar concentration greatly mitigates the concern about proton-electron damage equivalence. The observation that GaAs solar cells exhibit significant power output recovery from the effects of space radiation at temperatures that might be induced by certain concentrator designs provides an additional argument for its use. Currently three high effective concentration panel designs are being considered; the mini-Cassegrainian, SLATS and the magnesium Fresnel approach.

Until the challenges associated with GaAs cost and weight are met, concentration appears to be a very attractive vehicle for employing these cells for space applications. As the cost and weight of GaAs are reduced it will become a serious candidate for a wider range of future missions.

Second Generation Solar Cells

As mentioned previously, there is a strong emphasis on the part of DOE to develop high efficiency (18 percent AM1) solar cells for terrestrial applications. This broad based effort encompasses single crystal and amorphous silicon, thin films and cascade solar cells. The cascade technology is also being investigated by NASA and the Air Force for space applications. The technical resources being brought to bear on the high efficiency objective are quite impressive and there is a high probability that second generation terrestrial solar cell technology will provide some direct benefit to the space power program.

The cascade cell offers the best chance to achieve very high (greater than 25 percent) conversion efficiency. Cascade action in AlGaAs-GaAs structures has already been demonstrated. Theoretically a two junction device can approach 25 percent efficiency, but it is more likely that a three junction structure will be needed. There are a variety of technical issues associated with this technology.

The more apparent issues concern the method of forming the complex multi-layer structures, development of suitable acceptor impurity technology, and techniques for interconnecting the cascade stack. LPE has been used to fabricate two junction devices, but it is more likely that OM-CVD will be necessary for the more complex configurations. Although multiple beam epitaxy appears superior to OM-CVD with respect to layer chemistry control, current equipment does not lend itself to low cost and high volume processing. A number of techniques including the metal-interconnect, tunnel junction and germanium interconnect layer are being investigated as potential solutions for the cascade stack interconnecting problem.

Recent progress in improving the efficiency of amorphous silicon and thin film solar cells has been most impressive. Whether either device can ever achieve its efficiency goal (15-18 percent AM1) is moot. However, both cells have the potential for very high resistance to the effects of radiation because of their operating mechanisms. Thus it is possible that a 10 percent AMO amorphous silicon or thin film cell such as CuInSe_2 could offer improved end of life power, compared to conventional silicon and GaAs, for certain mission requirements.

CONCLUSION

The characteristics of the solar cell (efficiency, weight, radiation resistance and temperature coefficient) play a dominant role in determining the configuration of the solar array power generation system. No discussion of space photovoltaics would be complete without addressing the topic of solar array performance.

Two major figures of merit for solar arrays are specific power with respect to weight (W/kg) and area (W/m^2). The three major weight contributions to array weight are (1) the solar cell circuit (cell-cover-adhesive-interconnect), (2) the substrate upon which the circuit is mounted, and (3) the structure that deploys, supports and orients the array panels. Most arrays use rigid substrates which are relatively heavy, thus reducing the impact of cell weight. In the case of flexible (lightweight) substrate arrays, cell weight is much more important. However the effect of cell output per unit area is important for both types of arrays.

Figure 3 illustrates what has taken place during that last decade and projects what may occur in the next ten years. Since solar array performance is the ultimate manifestation of progress in photovoltaics, two arrays which have flown were selected. The FRUSA is the only U.S. flexible substrate array ever flown, while the FLTSATCOM design represents a typical example of the type of array now operating in space. To avoid having to make assumptions about structure weight, the actual array size was held constant for the calculations. The only change made was to substitute different cells, keeping all other circuit components (glass, interconnects, etc.) the same.

Three cells were chosen to illustrate the improvements made or anticipated; (1) a nominal ultrathin (75 μ m) silicon cell with a 28°C conversion efficiency of 13 percent, (2) a 200 μ m thick GaAs cell with 16 percent efficiency, and (3) an 18 percent GaAs cell whose effective weight is equal to that of the ultrathin silicon cell. Assembling losses, panel packing factors and temperature losses were held constant for each choice. In the case of GaAs it was assumed that the power loss caused by panel operating temperature was only half that of silicon.

Based on specific power (W/kg), it can be seen that in the case of the flexible array, GaAs cannot match the enhancement provided by ultrathin silicon until significant improvements are made in reducing thickness and increasing cell efficiency compared to that which is projected for the next five years. Rigid substrate panels do benefit from the implementation of near term GaAs technology, as was suggested previously. Dramatic improvements in both rigid and flexible solar array specific power are anticipated within the next ten years as the result of the development of an ultrathin GaAs cell which should possess a much greater efficiency than current devices due to further refinements in cell technology. Naturally the higher efficiency and reduced power losses due to panel operating temperature make near term GaAs cells very attractive for missions sensitive to the influence of solar array area. For both array types, panels composed of near term GaAs cells would be at least 25 percent smaller than those using ultrathin silicon cells.

End of life performance drives all array designs. The major factor in determining it is the predicted performance of the solar cell after exposure to the anticipated space radiation environment. The silicon solar cell is well characterized with respect to its behavior in both electron and proton radiation. Unfortunately GaAs has not yet matured to the state that its performance can be predicted with a high degree of confidence, therefore Figure 4 excludes GaAs.

Two orbits were selected, both at 30° inclination. One was at an altitude of 450 nmi (833 km), typical of many low earth orbit missions, and the other at geosynchronous, ~36000 km, used for defense and commercial applications. The coverglass thickness used in the original missions was retained and it was assumed that the flexible array provided only the same amount of backside shielding as the coverglass, while the rigid substrate array totally eliminated the influence of backside radiation. Mission duration was assumed to be 10 years.

The fact that the projected end of life specific power (W/kg) is greater than the original designs' beginning of life performance is convincing evidence that photovoltaics is still a vital technology that gives every indication that it has the capability to meet the challenge of future space power requirements.

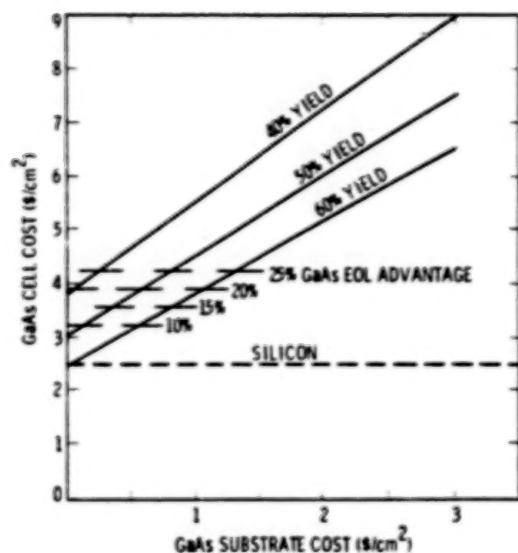


Figure 1. GaAs Cost Parameters - Planar Array

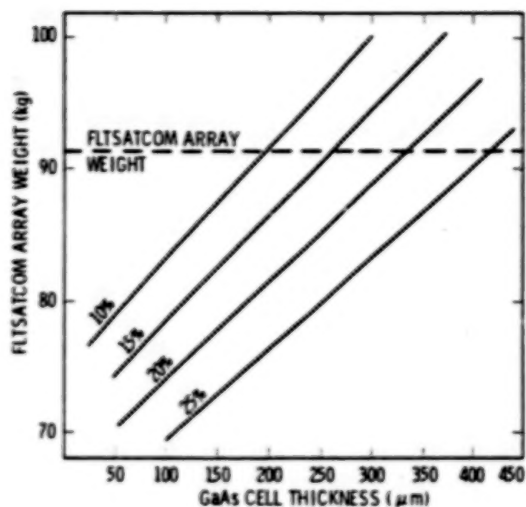


Figure 2. GaAs Weight Parameters - Planar Array

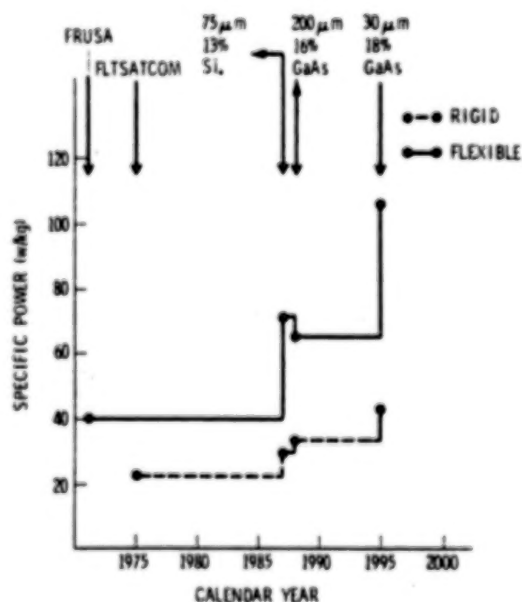


Figure 3. Projected BOL Planar Array Performance

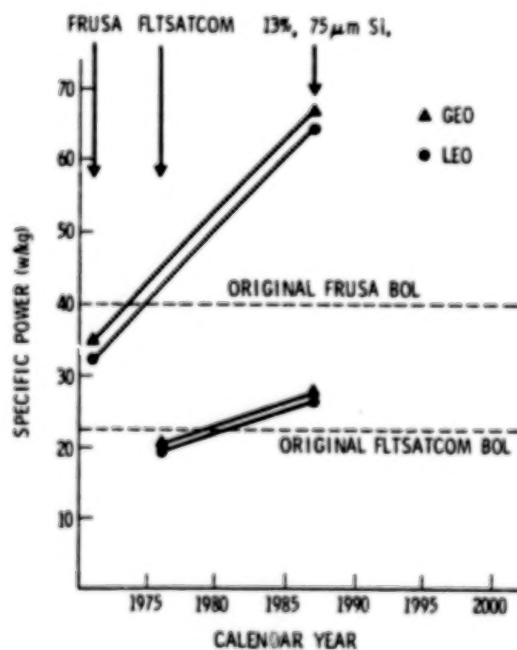


Figure 4. Projected (10 yr) EOL Planar Array Performance

ELECTROCHEMISTRY AND STORAGE

Lawrence H. Thaller
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The term "electrochemistry" implies the use of devices that convert chemical energy into electrical energy and sometimes vice versa. These devices are usually composed of some number of individual cells that are connected together to form a battery. In the cases where these devices cannot be electrically recharged they are usually referred to as primary batteries, whereas if these batteries can be charged and recharged repeatedly, they are called secondary batteries. Table 1 briefly lists the past and present uses of primary and secondary batteries in aerospace applications.

Secondary Batteries

Most of the spacecraft currently in use are powered by the hundreds of solar cells that are arranged into batteries (usually called strings) of the desired voltage level. During periods of darkness rechargeable batteries supply the power needs of the spacecraft. In low-earth-orbit applications a day/night cycle occurs about every 90 minutes on a very regular basis. In geosynchronous orbit there are two periods per year when darkness results from two eclipse periods that take place each year totaling about eighty days when there is some period of darkness. Over the years the energy storage requirements for spacecraft (NASA, DOD, & Commercial) have been met using nickel cadmium batteries. On rare occasions silver cadmium batteries have been used due to their special non-magnetic properties. The performance of batteries based on nickel cadmium has improved over the last twenty years or so even though the energy density (Whr/Kg) at the single cell level has remained almost constant. The term "performance" is a rather ill-defined term used to qualitatively describe parameters like "useable" energy density, cycle life, interval between reconditioning, etc. This improvement has resulted from attempts to establish good quality control procedures at the manufacturers' plants, as well as attempts to gain a more precise understanding of the basic electrochemical and chemical processes that take place within the cell. Also on a different front, a variety of more sophisticated battery system designs have been developed to more properly be able to deal with problems that arise from the cells having a certain degree of individuality. These designs incorporate more sophisticated electronic monitoring and control functions.

There has always been a desire to reduce the weight of the batteries used in space. These efforts have gone well beyond just trying to improve the nickel cadmium system. Cells and batteries that are based on more energetic couples than those used in the nickel cadmium system have been of interest for several decades. The mid-1960's was probably the high point in terms of trying to develop new battery systems to replace nickel cadmium. At the close of that era, work was begun on the nickel hydrogen system. Russian, German, and U. S. workers actively pursued this variant of the nickel cadmium system. At this point in time (1984) battery systems based on nickel hydrogen are beginning to displace those based on nickel cadmium for GEO applications. It should be noted that at the single cell level, the capacity and weight of an

individual cell of both types are about the same. The advantages of nickel hydrogen over nickel cadmium become evident at the battery system level where deeper depths of discharge (higher useable energy densities) are available from the nickel hydrogen system. Other systems that are currently receiving active attention are silver hydrogen, sodium sulfur, several types of non-aqueous lithium, and a rather recent entry into the field referred to as plastic batteries. Table 2 lists in some detail two general thrusts being taken in response to the requirements for higher power levels, higher voltage levels, higher energy densities, and longer service lives. Beside the development of cells and batteries employing higher energy density reactants, there has been a growing trend towards what will be called electrochemical systems. Efforts related to the development of bulk energy storage systems for terrestrial, as well as primary fuel cells for aerospace applications, were instrumental in advancing the technologies of large bipolar batteries where there is a commonality of reactants among all the cells of the battery. These system concepts allow a greater degree of freedom in design and operation compared to the batteries that are based on individual single cells. Figure 1 illustrates three different classes of batteries. On the top is illustrated the traditional type of battery where a cell string is placed in contact with a cold plate. In the center of that figure is illustrated a fully contained battery where an actively cooled stack of cells is employed. At the lower part of the figure a battery is illustrated where the actively cooled cell stack and the storage portions of the complete system may be treated somewhat independently. The stack of cells, complete with internal cooling passages, can be sized in terms of cell area and number of series connected cells to best meet the load requirements. In like manner, storage tanks for the electrochemical reactants are sized according to the requirements of the particular mission. The list of concepts on the lower right hand portion of Table 2 entitled "Electrochemical Systems" are those that are under some degree of consideration and development for future space missions which will be needing larger higher voltage storage subsystems. In this category, the hydrogen oxygen regenerative fuel cell is currently receiving the most attention followed by the bipolar nickel hydrogen battery effort.

Table 3 lists the major generic advantages that are associated with large actively cooled bipolar battery systems when compared with the more traditional battery concepts based on typically 50 Ahr single cells. Table 4 is an attempt to list the actual or projected energy densities of the major secondary battery concepts. They are divided between concepts that (based on their cycle life history) might never come under consideration for LEO applications where 30 to 50,000 cycles are desired and those that might come under consideration for GEO applications where only 80 cycles of varying depths of discharge are required each year. For the class of concepts that have been called electrochemical systems, the energy density numbers are a function of the orbit due to the different light to dark ratio and storage times.

The reader is referred to the proceedings of the IECEC, the Goddard Battery Workshop, the Power Sources Conference (both Brighton and Atlantic City) for further details and a more in-depth discussion of each of the concepts.

Primary Batteries

Primary batteries are devices that are filled with reactive material and after being discharged once, are then discarded. This is not always the case however, and furthermore, what is usually called a battery is really a cell and what is usually called a cell is really a battery. Probably the largest primary battery used in aerospace applications is the fuel cell system used on the space shuttle. There are about 6000 pounds of tanks, reactants, plumbing, and fuel cell stacks that can deliver about 2.5×10^6 Whr of power at about 30 volts DC. This works out to about 750 Whr/Kg for an energy density. It is very difficult to make small systems and still maintain this very high energy density. The individual unit devices placed in watches, calculators, and flashlights are more properly called cells. Over the last quarter century primary cells based on lithium have been developed by a number of laboratories to take advantage of the energetic properties of lithium. Primaries based on calcium have also been developed. Due to their reactivity with respect to water and even certain non-aqueous electrolytes, the commercialization of these devices has been slow. Several manufacturers are now available for the production of non-aqueous primaries for certain selected applications. For aerospace applications specific designs are currently under development. Energy densities in the range of 200 to 300 Whr/Kg are possible with these devices. Here again, the reader is referred to the technical literature and to specific manufacturers for further information on the characteristics and availability of primary cells for a particular application.

TABLE 1
ELECTROCHEMICAL ENERGY STORAGE

- 0 PAST APPLICATIONS
 - RECHARGEABLE NICKEL CADMIUM
 - RECHARGEABLE SILVER CADMIUM
 - PRIMARY SILVER ZINC
 - PRIMARY HYDROGEN OXYGEN FUEL CELLS
- 0 PRESENT APPLICATIONS
 - RECHARGEABLE NICKEL CADMIUM
 - RECHARGEABLE NICKEL HYDROGEN
 - PRIMARY SILVER ZINC
 - PRIMARY LITHIUM NON AQUEOUS
 - PRIMARY HYDROGEN OXYGEN FUEL CELLS

TABLE 2

CURRENT TRENDS IN ELECTROCHEMICAL ENERGY STORAGE

REQUIREMENTS	
0	HIGHER POWER LEVELS
0	HIGHER VOLTAGE LEVELS
0	HIGHER ENERGY DENSITIES
0	LONGER SERVICE LIVES
HIGHER ENERGY DENSITY COUPLES	ELECTROCHEMICAL SYSTEMS
0 NICKEL HYDROGEN	0 HYDROGEN OXYGEN REGENERATIVE FUEL CELL
0 SILVER HYDROGEN	0 BIPOLAR NICKEL HYDROGEN
0 SODIUM SULFUR	0 ZINC BROMINE FLOW BATTERY
0 LITHIUM (NON AQUEOUS)	0 HYDROGEN HALOGEN REGENERATIVE FUEL CELL
0 PLASTIC BATTERIES	0 HIGH TEMPERATURE SOLID OXIDE ELECTROLYTE
0 OTHER	0 OTHER

TABLE 3

**GENERIC ADVANTAGES OF LARGE, ACTIVELY COOLED,
BIPOLAR SYSTEMS(5-10 KW and larger)**

- 0 INCREASED GRAVIMETRIC AND VOLUMETRIC ENERGY DENSITY COMPARED TO THE EQUIVALENT DESIGN BASED ON SINGLE CELLS
- 0 HIGHER CELL VOLTAGES, ROUND TRIP EFFICIENCIES AND PEAK POWER CAPABILITIES
- 0 THE USE OF ACTIVE COOLING PERMITS SCALING FROM ONE SIZE TO ANOTHER WITH CONFIDENCE AS WELL AS A GREATER FLEXIBILITY IN POWER LEVELS
- 0 H_2-O_2 SYSTEMS HAVE THE ADDED POTENTIAL OF BEING INTEGRATED INTO THE PROPULSION AND LIFE SUPPORT SYSTEMS
- 0 SYSTEM REQUIRING HIGH VOLTAGES ARE EASIER TO DESIGN AND FABRICATE USING BIPOLAR DESIGNS

TABLE 4

PROJECTED ENERGY DENSITIES OF ELECTROCHEMICAL STORAGE DEVICES

BATTERY PACK TYPES		USEABLE SINGLE CELL ENERGY DENSITY - WHR/KG	SYSTEM TYPES	USEABLE SYSTEM ENERGY DENSITY - WHR/KG	
				LEO	GEO
Ni-Cd	25% DOD	10	Ni-H ₂ BIPOLAR	35	40
Ni-H ₂ IPV CPV	50% DOD	20	H ₂ -O ₂ RFC Eff Opt	20	60
			Wt Opt	30	90
Na-S 300°C	80% DOD*	100-150	H ₂ -Br ₂ RFC*	80	90
Ag-H ₂ IPV CPV*	50% DOD*	25-30	H ₂ -Cl ₂ RFC*	100	120
Na-X 200°C	80% DOD*	80-100	Zn-Br ₂ *	30-60	40-70
Li-FeS 400°C	80% DOD*	80-100			
Li-X Non Aq	80% DOD*	TBD			

*CURRENT CYCLE LIFE EXPERIENCE WOULD SUGGEST USE IN GEO ONLY

±PROJECTED LIFE APPEARS ATTRACTIVE BUT NOT ESTABLISHED

THREE DIFFERENT CLASSES OF ELECTROCHEMICAL STORAGE SYSTEMS

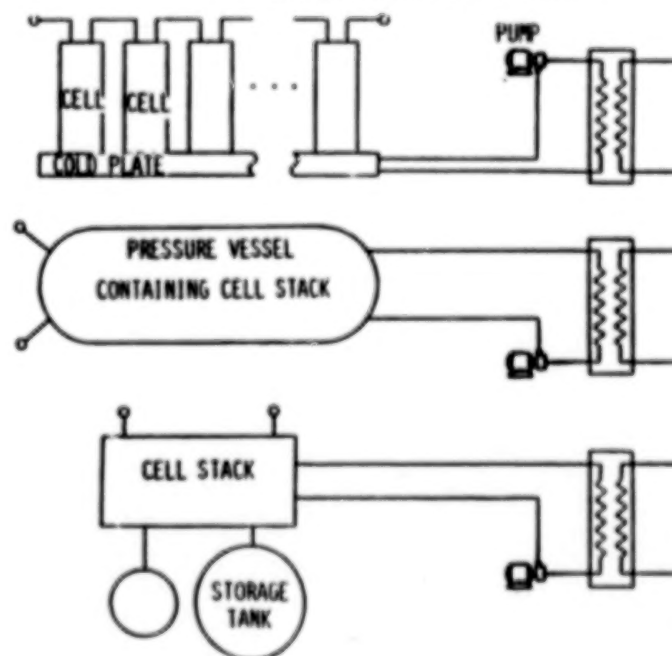


Figure 1.

POTENTIAL OF FLYWHEELS FOR SPACECRAFT ENERGY STORAGE

Sidney Gross
Boeing Aerospace Company

Energy storage systems for spacecraft in the past have used nickel cadmium and nickel-hydrogen batteries for rechargeable systems, or hydrogen-oxygen fuel cells for relatively short duration missions, such as Apollo or Shuttle. Regenerable fuel cells have also been evaluated and found to have good potential for space stations. Though flywheel systems have been suggested for spacecraft in past years, only recently have they been given serious consideration for spacecraft.

In the flywheel energy storage concept, energy is stored in the form of rotational kinetic energy using a spinning wheel. Energy is extracted from the flywheel using an attached electrical generator; energy is provided to the flywheel by a motor, which operates during sunlight using solar array power. The motor and the generator may or may not be the same device. Either magnetic bearings or mechanical bearings may be considered for flywheel systems.

Motor and generator functions can be satisfied either as two separate components, or integrated into one bifunctional component. Motors and generators are common in much of their design, and in some applications are interchangeable. Therefore, a combination motor/generator is lightest. A variety of arrangements of flywheels, motors, and generators is also possible (Figure 1).

FUNCTIONS OF FLYWHEEL SYSTEMS

The main function of a flywheel system is energy storage. Regulation of bus voltage is a valuable side benefit obtained from the generator, resulting in closer regulation than is obtained by either batteries or regenerative fuel cells. Other possible functions of flywheels systems (Figure 2) are attitude control and attitude reference. Attitude control systems can be postulated based on the momentum change of the flywheel, and attitude reference can be obtained by measuring torques exerted on a wheel.

FACTORS AFFECTING FLYWHEEL ENERGY DENSITY

High energy density is an important objective in spacecraft energy storage systems (Figure 3). The theoretical energy density obtainable is proportional to the maximum design stress of the material, and inversely proportional to material density. For isentropic materials, the ideal exponential disk (Stodola wheel) gives the maximum energy density. Other shapes (Figure 4) give less than ideal energy density. For anisentropic materials, such as composites, the energy density does not exceed 50 percent of theoretical (Figure 5).

Carbon fiber composite flywheels have a potential for high energy density (Figure 6). Also, these materials suffer very little from fatigue (Figure 7). Research on carbon fibers is making significant progress, and forecasts of improved materials are very encouraging (Figure 8).

Composite flywheels with circumferential fibers lack strength in the radial direction. Unless special design and manufacturing steps are taken to compensate for this, a carbon-

epoxy flywheel will have its highest stress at the inside of the rim (Figure 9). A failure there would progress outward, often breaking the wheel into three pieces with a large amount of energy and momentum to be dissipated or transferred, thus creating a serious containment problem. A promising approach is the use of a urethane elastomer instead of epoxy. Analyses show that this will result in the highest stresses at the outside edge of the rim (Figure 10). This is preferred, for it results in circumferential rupture of the outermost fibers, whereby only minor fragmentation is released; failure can be detected and the wheel can be shut down with minimum damage and energy release. Experimental verification of this concept is needed.

EFFICIENCY CONSIDERATIONS

Energy storage efficiency is a key factor in the optimization of spacecraft energy storage systems, and also in the choice between one system and another. The problems of large sized solar arrays are well out of proportion to their modest weight, and an efficient energy storage system reduces the size of the solar array. This is shown parametrically in Figure 11. For high-power spacecraft with large solar arrays, significant quantities of propulsion fuel must be resupplied regularly to offset the effects of solar array drag, and maintain the spacecraft within the selected orbit. Inefficient energy storage systems require greater solar area, and hence more propulsion fuel. This is shown in Figure 12 for a typical space station design using either hydrazine or hydrogen-oxygen propellants. This penalty can be considerable over the life of the spacecraft.

The calculated efficiency of the flywheel energy storage system is shown in Figure 13. For the intermediate design objective, the overall efficiency is 81.1 percent; for the advanced design objective, the overall efficiency is 92.8 percent. Motor/generator efficiency is the major contributor to losses in both cases. Electrochemical systems, by comparison, are on the order of 55 to 65 percent efficient.

WEIGHT COMPARISON

Motor/generator weights required will be related to flywheel system speed (Figure 14). Typically, systems would be designed for about a two-to-one ratio of maximum-to-minimum wheel speed. Efficiency of the motor/generator is also related to speed (Figure 15).

Flywheel energy storage system weights are shown in Figure 16. It is seen that flywheels are lighter than the battery systems when comparisons are made at the design depth-of-discharge, for the flywheel can cycle repetitively at deeper depths-of-discharge than can batteries. This can be a valid comparison only if the reserve capacity of the battery systems is not depended upon for emergency power. The flywheel system is not practical for depths-of-discharge much greater than 75%, and the upper practical limit for battery systems for occasional discharges is approximately 85% depth-of-discharge for nickel-hydrogen, and 75% for nickel-cadmium batteries. A weight comparison for these design values is given in Figure 17, applicable for comparison of emergency power capability. Even for this condition the flywheel system is lightest.

Typical weight comparisons at the spacecraft level have been made between flywheel systems, regenerative fuel cell systems, and battery systems (Figure 18), all with equal emergency power capability. The power system load is 50 kw for both sunlight and occultation in low earth orbit. The higher efficiency of the flywheel system accounts for an important part of the weight saving, for less propellant resupply is required due to the smaller solar array needed. Lower propellant resupply over a period of many years can be a major advantage of the flywheel system.

VOLTAGE RANGE EFFECTS

An inherent characteristic of secondary batteries is a relatively wide bus voltage spread due to the large difference between charge and discharge voltage. A regenerative fuel cell system will have about half the voltage spread of a Ni-H₂ battery. A flywheel generator, on the other hand, will control voltages very closely, within approximately two percent. This not only makes the design of internal power supplies lighter and more efficient, but also eliminates one of the major problems of spacecraft power systems. An estimate of the typical improvement in efficiency of these loads is shown in Figure 19. It is seen that most of the loads could be reduced 0.8 percent using the tighter voltage regulation obtainable with a motor/generator. Non-essential loads, such as payloads, could probably take advantage of the potential saving. However, loads essential to the operation of the spacecraft probably would have to be designed to meet the expected wide voltage range of the launch power source and the emergency batteries, and therefore could not take advantage of this.

UTILIZATION OF EXCESS SUNRISE POWER

Spacecraft solar arrays become cold during occultation. Upon emergence into the sunlight, there is a higher voltage output, hence a high power output. This increased power condition lasts for about 20 minutes, depending on the time to reach steady sunlit temperature, which is determined mostly by the unit thermal mass of the solar array. Typical solar array performance in low earth orbit is shown in Figure 20. The incremental power due to the low temperature transient is seen to be an increase in solar array output of approximately seven percent. This potential for extra power usually is not used, as for example in a shunt regulated power system. In the less common series regulated system with pulse-width modulated control, part of the excess power is sometimes used for battery charging, but this can compromise the batteries, which are charge-rate sensitive. Flywheels, within limits, are not charge-rate sensitive, and thus can make use of this additional power.

LIFE AND RELIABILITY

Life and reliability of nickel cadmium batteries are important concerns for all spacecraft applications, including the space station. Nickel hydrogen batteries have the potential for improved life and reliability, and efforts are now being expended to develop that potential. For either system, however, it is expected that periodic battery replacement will be necessary to meet the space station lifetime requirements.

Flywheel systems have the capability for much longer lifetimes than do battery systems; when developed, the flywheel system should be able to operate without replacement during the life of a space station, in the range of 10 to 30 years. In assessing the life and reliability of the flywheel motor/generator system, those items considered to be key to long life and reliability are: (1) fatigue and long term creep of the flywheel rotor; (2) bearings; (3) control electronics; (4) cooling system. Flywheel system lifetime probably is limited by the associated electronics, which can be designed to be replaceable.

Magnetic bearings offer the most promise for long life spacecraft applications. These need involve no mechanical contact between the rotating equipment and the stationary elements. Degradation of the permanent magnet elements in the bearings is expected to be minor over 15 years. Thus, the electronics required for the magnetic bearing control may be the critical long life item for the bearings.

CONCLUSIONS

Flywheel energy storage systems have good potential for use in spacecraft such as the space station (Figure 21). This system can be superior to alkaline secondary batteries and regenerable fuel cells in most of the areas that are important in spacecraft applications. Of

special importance, relative to batteries, are lighter weight, longer cycle and operating life, and high efficiency which minimizes solar array size and the amount of orbital makeup fuel required. In addition, flywheel systems have a long shelf life, give a precise state of charge indication, have modest thermal control needs, are capable of multiple discharges per orbit, have simple ground handling needs, and have characteristics which would be useful for military applications.

The major disadvantages of flywheel energy storage systems are that power is not available during the launch phase without special provisions; and in-flight failure of units may force shutdown of good counter-rotating units, amplifying the effects of failure and limiting power distribution system options. Additional disadvantages are: no inherent emergency power capability unless specifically designed for, and a high level of complexity compared with batteries. In net balance, the potential advantages of the flywheel energy storage system far outweigh the disadvantages.

The major technology issues with flywheel systems (Figure 22) are: rotor design; rotor containment; motor/generator design; and system level issues. Further analysis, research, and development are required on these items.

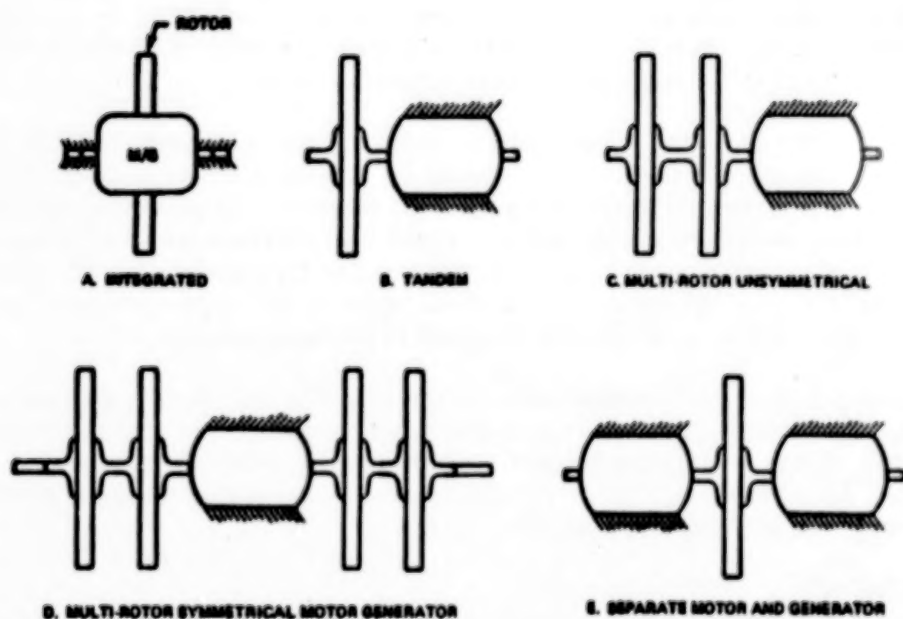


Figure 1. - Motor/generator - rotor design approaches.

- Energy storage
- Bus voltage regulation
- Attitude control (torquing)
- Attitude reference (measure torques exerted on wheel)

Figure 2. - Possible functions performed by flywheel system.

- Wheel shape
- Tensile strength/density ratio
- Isentropic/anisentropic properties
- Design/manufacture for low radial strength compensation
- Hub design

Figure 3. - Factors affecting flywheel energy density.




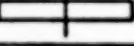
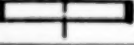
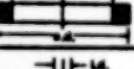


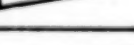

IDEAL EXPONENTIAL DISK		1.0
TRUNCATED EXPONENTIAL DISK		0.807 (rhub) = 0.48 (rtip)
TRUNCATED CONICAL DISK		0.808
SOLID FLAT DISK		0.808
IDEAL THIN RING		0.800
THICK RIM		0.438 $\frac{L}{A} = 0.1$ $\frac{R}{A} = 0.1$
PIERCED DISK		0.308 $\frac{L}{A} = 0.1$ $\frac{R}{A} = 0.1$
WHEEL WITH SPOKES		0.4
THIN BAR		0.333
SHAPED BAR		0.5

Figure 4. - Shape factors for some isotropic flywheel rotor designs.

DESCRIPTION	SHAPE FACTOR K
CIRCUMFERENTIALLY WRAPPED FLAT DISK	0.43
CIRCUMFERENTIALLY WRAPPED FLARED DISK	0.47
TAPE OVERWRAP	0.35
MULTI-RIM	0.45

Figure 5. - Shape factors for some composite material rotor designs.

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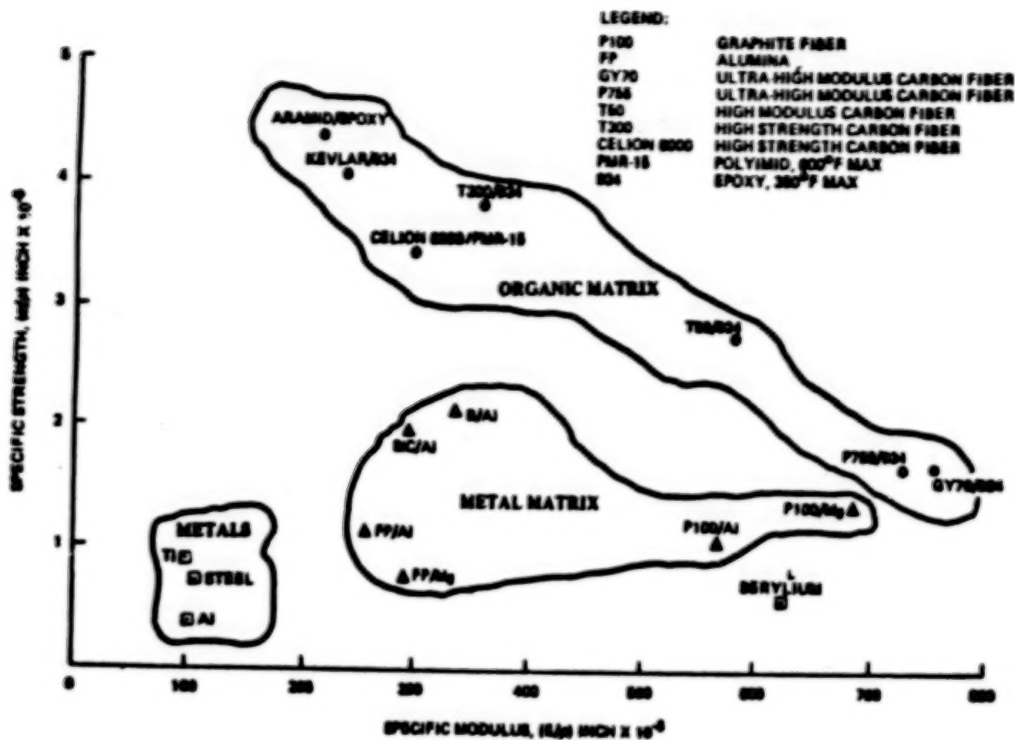


Figure 6. - Strength of composite materials.

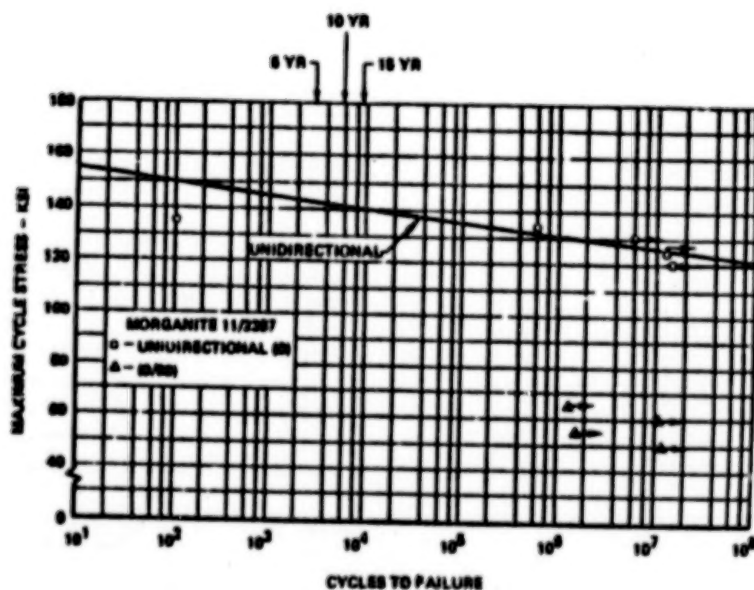


Figure 7. - Constant amplitude unidirectional fatigue properties of typical high-strength graphite/epoxy composite.

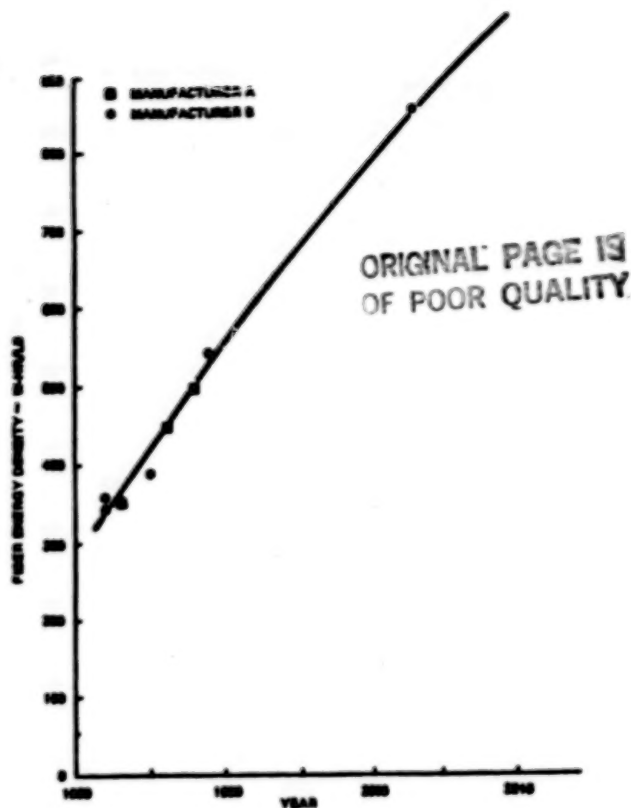


Figure 8. - Energy density from projected development of high-strength carbon fibers for 1980-2010.

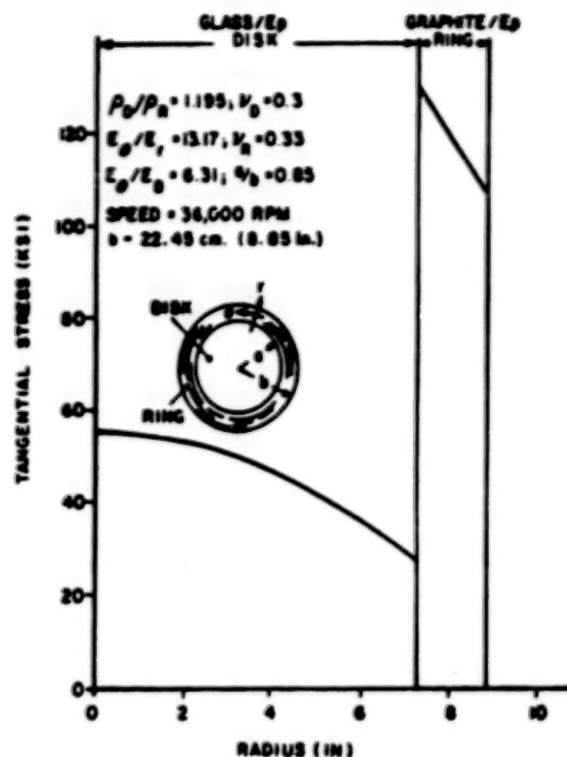


Figure 9. - Stress distribution with epoxy elastomer.

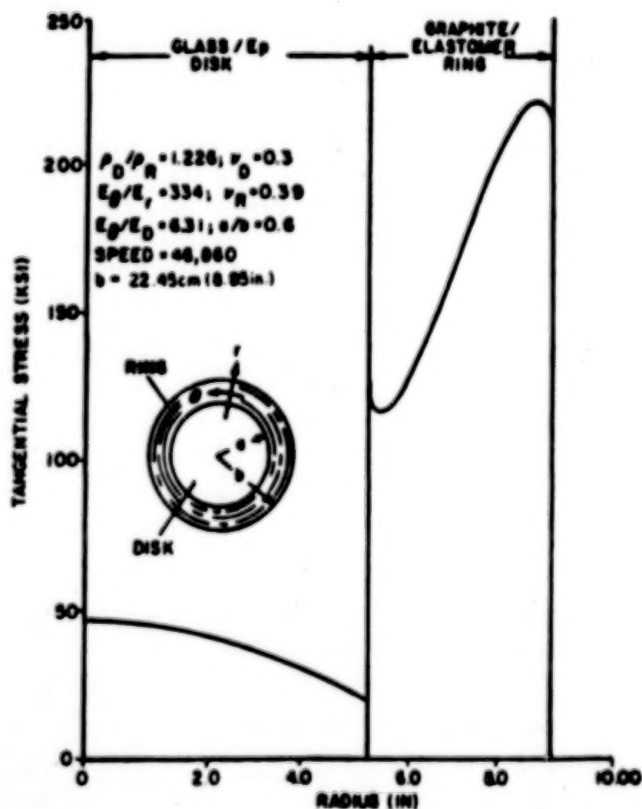


Figure 10. - Stress distribution with urethane elastomer.

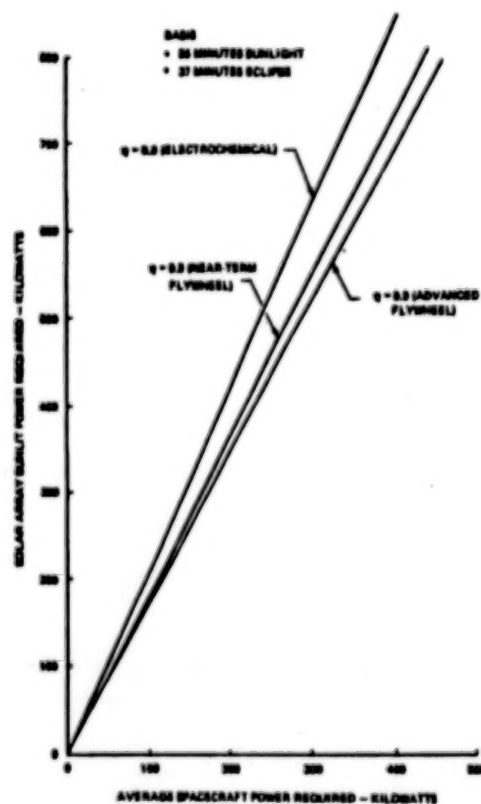


Figure 11. - High-efficiency energy storage systems result in smaller solar arrays.

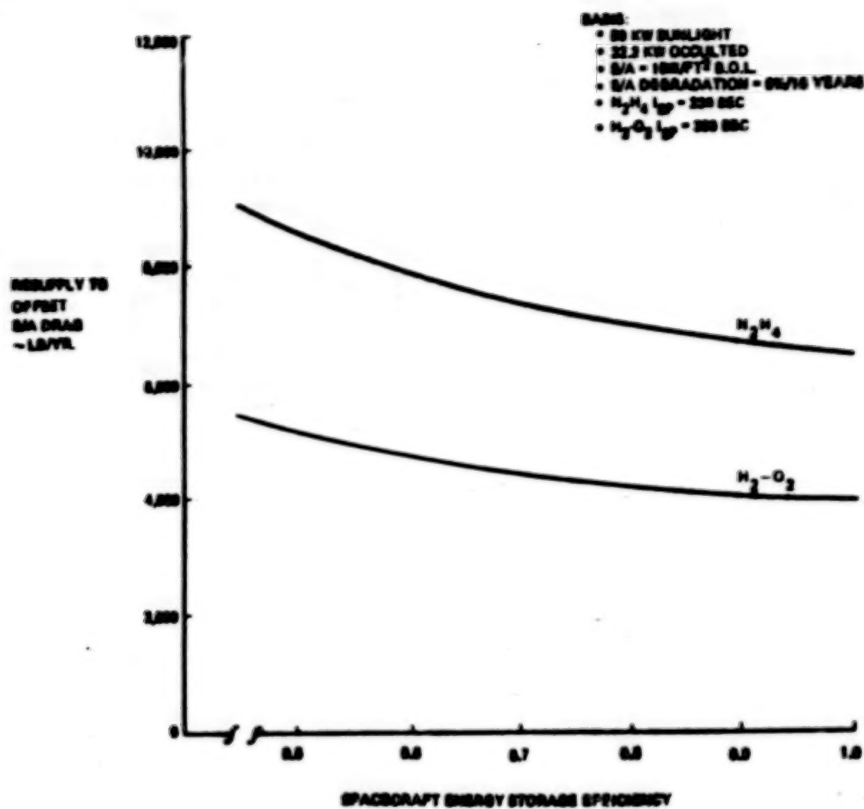


Figure 12. - Propulsion resupply due to solar array drag.

	EFFICIENCY	
	INTERMEDIATE DESIGN OBJECTIVE	ADVANCED DESIGN OBJECTIVE
LOSSES FROM CYCLIC STRESS	100%	100%
MOTOR EFFICIENCY	89.5%	96.5%
GENERATOR EFFICIENCY	91.0%	96.5%
SOLAR ARRAY CHARGE AREA EFF.	100%	100%
HEAT PIPE POWER	100%	100%
MAGNETIC BEARING POWER	99.61%	99.72%
OVERALL EFFICIENCY	81.1%	92.8%

Figure 13. - Energy storage efficiency with flywheels.

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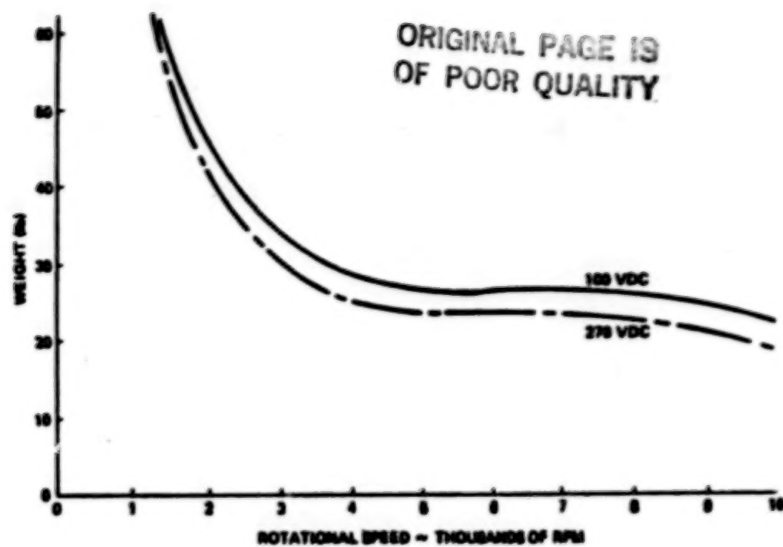


Figure 14. - Weight of conventional design motor/generators and controls.

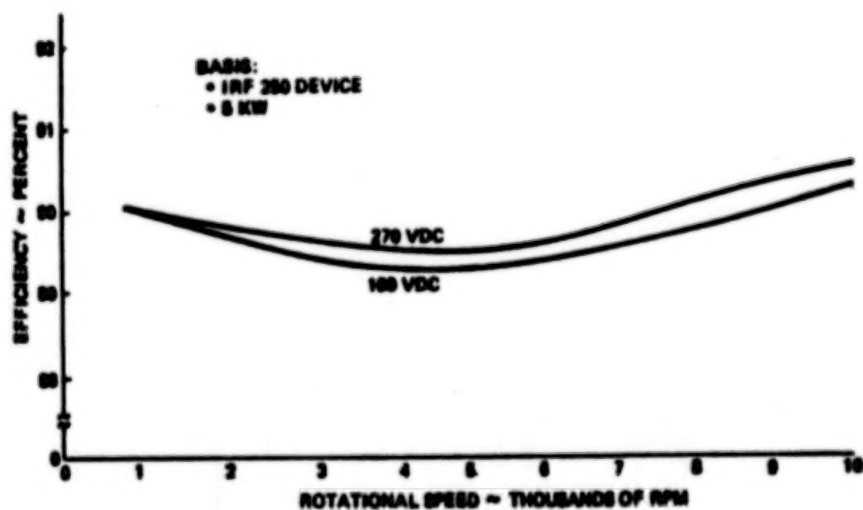


Figure 15. - Efficiency of conventional design motor/generators.

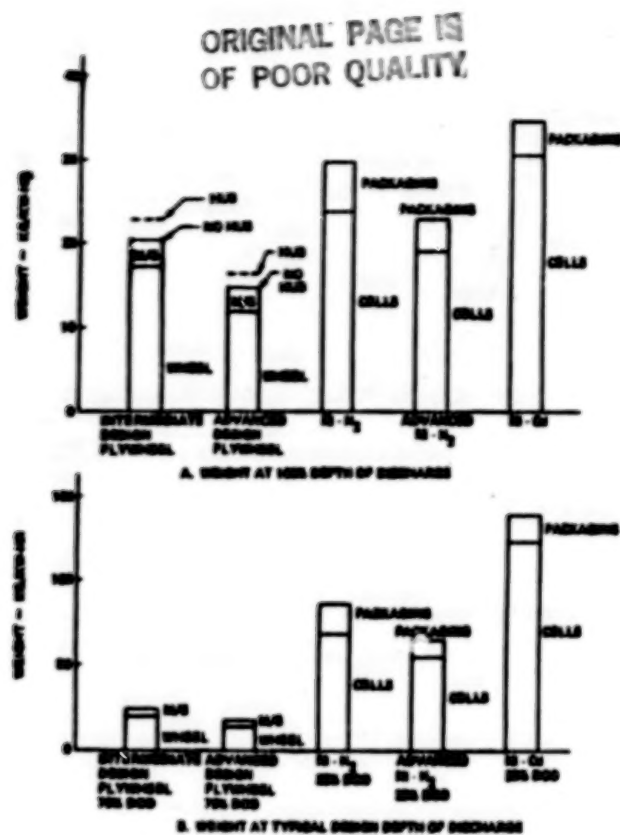


Figure 16. - Comparative weights of energy storage devices.

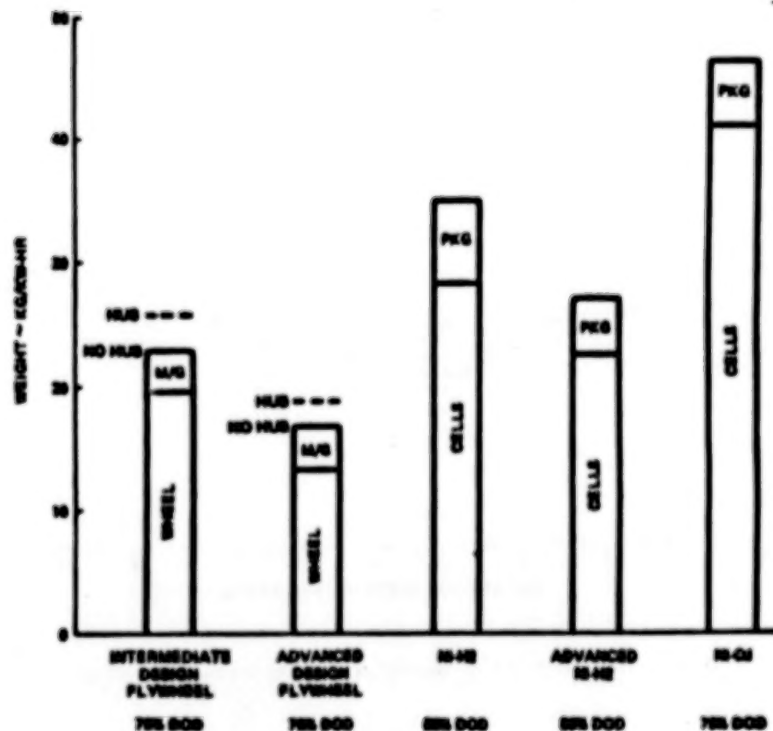


Figure 17. - Comparative weights of energy storage devices sized for emergency condition.

	Ni-Cd 25% DOD	Ni-H ₂ 35% DOD	Regenerative Fuel Cells, Efficiency Optimized	Flywheel with Emergency Capability	
				Intermediate Design Objective	Advanced Design Objective
Energy Storage	8,769	5,796	2,545	4,931	3,564
One Year Total Weight, lbs	17,220	14,321	9,466	10,893	9,010
Ten Year Total Weight, lbs	46,749	45,272	38,995	36,354	33,022

Figure 18. - Weight summary.

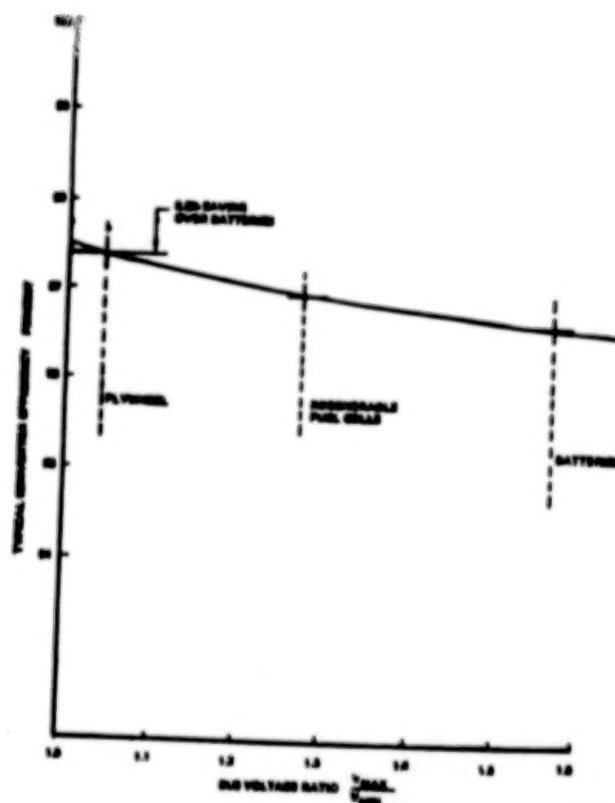


Figure 19. - Effect of bus volt regulation on power system efficiency.

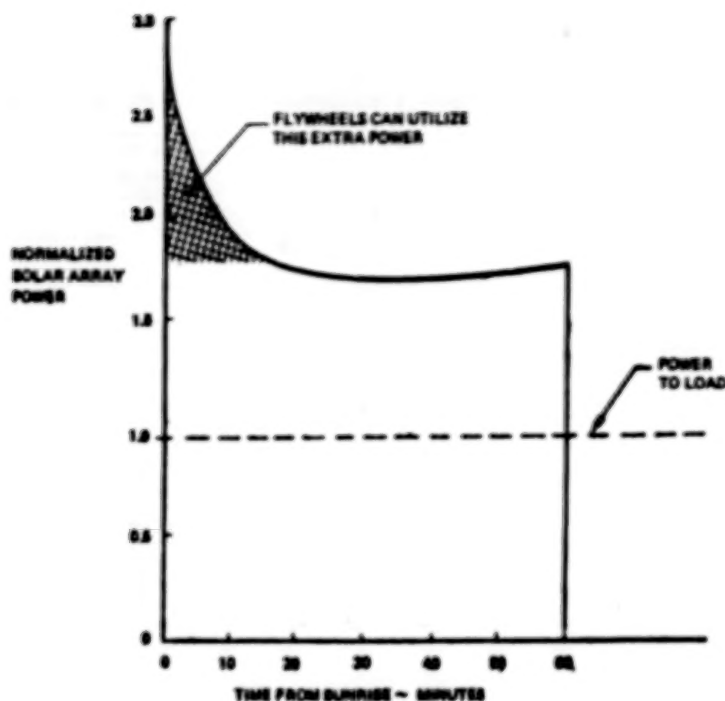


Figure 20. - Typical Sun-oriented solar array performance in low Earth orbit.

Flywheels can be better than batteries and fuel cells

- Low weight
- Long life
- High efficiency
- Smaller solar array
- Less orbital make-up fuel
- Long shelf life
- State of chg indication
- Easy thermal control
- Can use excess sunrise power
- Multiple discharge capability per orbit

Flywheels have some disadvantages

- Power not available during launch
- Failure forces shutdown of counter-rotating unit
- No inherent emergency power capability
- System is complex

Much R&D needed

- Rotor technology
- Magnetic Bearings
- Rotor containment
- Efficient motor/generators
- Hub technology

Figure 21. - Conclusions.

- **Motor/Generator**

- Pick best flywheel/motor-generator speeds
- Design for high efficiency
- Design for wide speed range
- Design for tight voltage regulation
- Design to use excess sunrise power
- Determine if mechanical bearings have any application
- Design magnetic bearings for flywheel/motor-generator combination

- **Rotor Design**

- High strength, high energy density
- Design/manufacture to compensate for low radial strength
- Design for fail-safe rupture
- Hub design is difficult

- **Rotor Containment**

- Give protection from failure
- Manage debris
- Provide vacuum shell for ground test

- **System Level Issues**

- Impact of integration with attitude control system
- Configuration optimization
- Control – electrical, suspension, momentum
- Analytical modeling needed
- Operation following unit failure
- Data base for systems and components
- Applicability to GEO & LEO
- Applicability to small and large spacecraft
- Safety
- Cost and schedule

Figure 22. - Major technology issues.

TRENDS IN CONCENTRATOR, RECEIVER, AND STORAGE DEVELOPMENT
FOR TERRESTRIAL APPLICATIONS

E. John Roschke, William A. Owen, Wesley A. Menard, and
Toshio Fujita
Jet Propulsion Laboratory

The experience gained and the technical trends of the DOE-sponsored terrestrial solar Thermal Power Systems project are summarized with respect to concentrator and receiver/storage development. Relevance of this experience to space power applications, and the perceived critical barriers of this technology, are discussed. It is concluded that, despite different objectives, the terrestrial program provides a strong basis of expertise that will be valuable to space power applications development.

INTRODUCTION

The Jet Propulsion Laboratory was appointed by NASA as manager of the Department of Energy's solar parabolic dish program in 1977. Three major thrusts were developed in parallel. The first was a technical development program to assess the state of the technology, particularly solar concentrators, high temperature receivers, small (20 kWe) heat engines, power generation and conditioning, and controls. Suitable equipment was developed with the help of industry. From this came the Test Bed Concentrators, Parabolic Dish Concentrator No. 1, high temperature receivers for Stirling, Brayton, and Rankine (both organic and steam) systems, adaptation of existing heat engine programs to solar needs, numerous testing and development techniques, control and data acquisition systems, and all ancillary equipment for safe and efficient power conversion systems.

The second thrust was to combine these elements into autonomous, sun-tracking modules each typically producing 20 to 25 kW of electricity. This included not only a suitably matching concentrator, receiver and engine/generator but all the necessary controls and power conditioning equipment to demonstrate the utility of the module. Baseline data was gathered on efficiencies, maintenance, operations, manufacturing methods, installation costs, and all other elements necessary to establish realistic life-cycle costs. Building on several existing automotive and industrial power programs, where large dollar efforts were in progress, a number of modules were implemented. These efforts included the kinematic Stirling, an organic Rankine, a regenerated Brayton, and a small (8 kW) subatmospheric Brayton module. All of these units are in various stages of test.

The third thrust was to deploy limited capacity plants consisting of individual modules in actual field situations to assess how this new technology would impact the existing power industry. This phase is currently in progress. Throughout the course of the effort, an aggressive in-house research and development effort was pursued. This effort resulted in an extensive subsystem development and subsequent

field testing experience. However, to date, extensive field testing of complete, generic system modules for electric power generation has not been accomplished. The earliest such module, the Vanguard Stirling module designed by the Advanco Corp., has been installed at Rancho Mirage, California, and is undergoing preliminary testing.

Goals for the DOE/JPL effort were to (1) develop first and second generation systems and components to meet specified cost and performance targets for 20 to 30 year lifetimes, (2) reduce life cycle and energy costs through large-scale production techniques to enable the development of competitive modules and plants, and (3) transfer technology to industry. In 1983, expanding commitments to space activities forced JPL management to terminate this effort and, subsequently, DOE selected Sandia National Laboratory at Albuquerque, New Mexico (SNLA) to continue the program. An orderly transition from JPL to SNLA is in progress. This includes a phased shutdown of JPL's Parabolic Dish Test Site (PDTS) located at Edwards Air Force Base, California, and transfer of equipment to SNLA.

This paper summarizes the technical development that has occurred within the Thermal Power Systems (TPS) terrestrial program in the area of solar concentrators, receivers, and storage units. Status, trends, and performance levels are described along with field operations and test evaluation methods. Relevancy of the terrestrial technology to space power applications and the perceived critical barriers are discussed briefly. To save space, no hardware photos have been included. Readers desiring such information are referred to reference 1, which should be available to many of the attendees at this Space Power Workshop.

SYSTEM CONSIDERATIONS

The concentrator, receiver (together called the collector), and power conversion unit assembled together make up an individual module for generation of electric power (fig. 1). Thermal storage may be provided as an integral part of the receiver or by an external device and thermal transport. Buffer storage (a few minutes) or long-term storage (an hour or longer) are provided to meet the requirements of anticipated insolation interruptions. Electric storage may be used as an alternative, or together with thermal storage.

The concentrator is classified as point-focusing when the solar image is focused in an intense localized spot, and may be most easily visualized as a circular paraboloidal dish-type mirror, which collects and focuses the incoming solar energy as it tracks the sun. At the focal point of the concentrator, the focused solar energy passes through the aperture of the receiver into its cavity, which may be cylindrical or have other configurations. In the cavity, the solar energy is absorbed and transferred to a working fluid that transports heat to the engine. Cavity receivers have become conventional but other types, e.g., external, are possible. The receiver and entire power conversion assembly may be mounted at the focal point but ground-mounted engines/alternators have been considered as well.

Numerous systems analyses and trade studies are available, e.g., reference 2. There is a tradeoff between optical efficiency (ratio of energy passing through the aperture to energy incident on the concentrator) and geometric concentration ratio (ratio of aperture area to concentrator projected area) through the intercept factor (ratio of energy intercepted by the aperture to the energy incident on the infinite

focal plane). For a given concentrator, the intercept factor increases with increasing aperture size and yields increasing optical efficiency. But, larger aperture energy losses result as a consequence of larger apertures. In general, intercept factors greater than 0.9 are desirable when high receiver temperatures are required.

The geometric concentration ratio needed to provide reasonable collector performance increases with receiver temperature. Small receiver apertures are desirable at high temperatures where re-radiation becomes very important. Collector efficiency as a function of receiver temperature is shown for a typical case in figure 2 for various geometric concentration ratios. Figure 3 shows system efficiency as a function of receiver temperature for various engine characteristics. The engine selection will influence greatly the choice of receiver temperature and, hence, the geometric concentration required.

The performance of a concentrator is governed by its geometrical configuration and critical thermo-optical properties, including solar reflectance, specular spreading due to microscopic roughness, surface error, and tracking/pointing accuracy. Solar reflectance and specularity are governed by mirror material selection. Three basic categories of reflector material are generally considered for solar concentrator applications: (1) second surface mirror, (2) metalized plastic films, and (3) polished metal surfaces. Surface degradation and dirt build-up on the reflector surface are very significant factors governing the reflectance properties. The effect of reflectance on the system performance is linear. The specularity, however, plays only a minor role for most reflectance materials.

In general, the geometrical center of the collector/receiver does not coincide with the center of the solar image due to the concentrator pointing error. The pointing error includes errors due to inaccurate sun tracking, misalignment, and structural deflections caused by gravity and wind loads.

Surface slope error has been identified as the most significant parameter governing the optical performance of a solar concentrator. An ideal concentrator would have the reflector surface contoured precisely to the shape required by geometrical relationships. However, shape deviations may be caused by macroscopic surface waviness, imperfect alignment, and slope errors due to manufacturing tolerance and structural deflections. The local surface error of a reflector element is defined as the angular deviation of the surface normal from that of a perfect geometry. Theoretically, a detailed mapping of surface error over the entire concentrator body would give the most accurate description of the surface error. The value may vary considerably from center to rim, and from zone to zone, circumferentially. From a practical point-of-view, the surface errors have to be characterized in a statistical manner. A standard sampling technique must be developed to establish the effective surface error statistics through quantitative measurements.

Receiver performance depends on many factors such as operating temperature, aperture size, cavity geometry, optical properties of absorbing surfaces, alignment and positioning, spillage, heat exchanger characteristics and heat loss mechanisms. Methods for improving receiver efficiency are discussed in reference 3.

CONCENTRATOR DEVELOPMENT

An extensive review of recent terrestrial concentrator development is provided in reference 4. Table 1 (adapted from ref. 4) lists general information, design and construction features, and performance values for selected concentrators; table 1 includes just seven of the twenty-two entries listed in reference 4. Two Test Bed Concentrators (TBCs) and a Parabolic Dish Concentrator (PDC-1), a \odot design, were fabricated, installed and tested at the JPL Parabolic Dish Test Site (PDTS). Design characteristics of these concentrators are given in table 2, as well as in table 1.

The E-Systems Test Bed Concentrators (TBCs) were developed as an early tool for use in the solar energy development program to provide a precise, consistent, and highly reliable source of thermal solar energy for testing a variety of receiver and/or power conversion subsystems. The two TBCs have been operational at the PDTS since October 1979. They have a plan form diameter of nominally 11 meters, are parabolic in shape with a reflector having 224 JPL-developed, rectangular shaped, second surface, back silvered, long radius, spherical contoured mirrors (ref. 5). Each mirror facet is individually aligned. The concentrators are of the Elevation over Azimuth tracking type with an azimuth wheel and track design, and a jack screw elevation drive. The sun sensor/control loop keeps the concentrators pointed to within 0.05° of the sun's true position. Reference 6 reports early characterization studies.

The test data has substantiated that the TBCs have fulfilled their design purpose by providing flux densities well in excess of those required for nominal testing sequences. Figure 4 shows a computer-generated focal plane flux distribution based on early data taken with 100 percent of the mirrors uncovered. In fact, the peak fluxes measured with the initial mirror alignment have been purposely reduced by defocusing a part of the central mirror facets. This was done in order to minimize thermal damage to the TBC receiver mounting structure and the receiver components. The defocusing did not significantly reduce the overall available energy even though the peak flux is down almost threefold. Steps in defocusing are shown in figure 5. For most receiver testing to date, the center mirrors were defocused to reduce peak flux intensity to about 600 W/cm^2 . Figure 6 shows how the flux distribution varies in front and behind the focal plane when the center mirrors are defocused.

The TBCs can produce a maximum of 82 kW thermal with an insolation of 1000 W/m^2 . With the mirror facets set for sharpest focus, a peak flux greater than 1500 W/cm^2 is obtained with an aperture diameter of 20 cm (7.8 in). Lower power levels can be obtained by covering a portion of the mirrors.

The \odot concentrator, called the PDC-1 (see table 2), has been constructed and was installed at the PDTS by Ford Aerospace Communications Corporation. The results of early optical testing are reported in reference 7.

Acurex Corporation, under contract to the Jet Propulsion Laboratory, has developed a second generation point focusing solar concentrator concept. The design concept is based on the use of reflective gores fabricated from thin glass mirrors bonded continuously to a contoured substrate of cellular glass. The detailed design effort is complete; the concentrator aperture and structural stiffness has been optimized for minimum concentrator cost given the performance requirement of delivering 56 kW_{th} to a 22 cm (8.7 in) diameter receiver aperture with a direct nor

mal insolation of 845 watts/m^2 and an operating wind of 50 kmph (31 mph). The reflective panel, support structure, drives, foundation, instrumentation, and control subsystem designs, optimized for minimum cost, have been developed. The use of cellular glass as a reflective panel substrate material offers significant weight and cost advantages compared to existing technology materials. This concentrator is designated PDC-2.

Many concentrator design concepts have been investigated. Designs with secondary mirrors, such as the Cassegrainian type (which may have advantages for space applications) have been studied theoretically, but none have been built or tested. However, non-imaging secondary concentrators have been built and tested on a TBC (ref. 8). Preliminary results were encouraging; in a low power test, the amount of energy redirected into the aperture increased by more than 30 percent over the value obtained without the secondary concentrator.

RECEIVER DEVELOPMENT

Solar receivers are the link between the concentrated solar energy and the engine or process that utilizes the energy. While much time and effort have been expended on developing concentrators and heat engines, comparatively little has been spent on receivers. This is probably due to the perception that they are inherently simple, low cost devices. Recent system studies, however, emphasize that receivers play just as important a role in system efficiency as the more complex components.

Until recently, receivers were designed using conventional heat exchanger techniques. But when these designs were converted into hardware, none performed as well as expected with losses exceeding calculations by 5% to 50%. In retrospect, these often substantial differences are not surprising when the complexity of the receiver as a thermal system is assessed. In any complex system, analysis is difficult especially in finding omissions in the model, but this was especially true for receivers, which had been given little overall system analysis. And, too, very little previous work, either analytical or experimental, had been done on phenomena especially important to small cavities, such as aperture convection or gray body radiation.

As more point-focusing systems were constructed, a considerable body of data emerged. Examination of this data highlighted many of the special problems, especially for higher temperature systems. It became clear that a number of design aspects including cavity shape, use of windows, coatings, surface condition, radiative properties, cavity convection effects, reflection, wind screens, lifetime, and other more secondary characteristics needed integration into a comprehensive design scheme.

Receivers are identified with the heat engine cycle to which they provide thermal input. Also, they can be classified according to the working fluid they use, or by whether or not they contain phase change materials for buffer storage. A summary of characteristics of cavity-type receivers tested for organic and steam Rankine, Brayton, and Stirling cycles is presented in table 3. Not listed are several United Stirling of Sweden (USS) designs that were tested by USS personnel at the PDTS. The buffer storage indicated in table 3 for the Sanders Associates Brayton receiver consists of a ceramic mullite matrix; the heat exchange surface consists of a sintered beta silicon carbide honeycomb matrix.

A drawing of the AiResearch Brayton air receiver is shown in figure 7. Figures 8, 9, and 10 are included to display the results of typical analyses that are required to design a cavity receiver; all are taken from reference 9. The flux distribution and the temperature distribution along the cavity walls are shown in figures 8 and 9, respectively. Theoretical cavity efficiency as a function of thermal input energy for three different aperture sizes is shown in figure 10. As noted earlier, experimental receiver efficiencies tend to fall well below theoretical values. This is evident by comparing the results of figure 10 with the experimental values for the Brayton receiver given in table 3.

In figure 11 is shown early experimental data obtained with the Ford organic Rankine receiver. Temperature and pressure history can be correlated with insolation, as the insolation becomes interrupted by a series of clouds.

STORAGE DEVELOPMENT

Thermal storage for terrestrial solar thermal systems is most cost effective when its application is limited to buffer storage (fractions of an hour) as required to maintain power output during intermittent cloud passage. Early contractor conceptual design studies indicated that the cost and weight of cavity receivers was approximately doubled as the result of incorporating about 10 min of buffer storage. However, buffer storage for dish systems can be incorporated effectively into the receiver, as shown by studies conducted by GE and FACC, and by JPL (ref. 10). Reference 11 contains a thorough study of thermal and electric storage.

The selection of storage materials depends on temperature level, physical properties, energy density, cost, availability, and fabrication and containment difficulties. Latent heat materials are likely to be the best choice for terrestrial thermal storage applications. Characteristics of some candidates for high-temperature latent heat storage materials are given in table 4.

Long-term storage concepts have been developed (ref. 12) but none have been fabricated to completion. In the GE heat pipe receiver for powering a Stirling engine (fig. 12), sodium conveys heat to a phase-change storage section containing salts that surround the heat pipes. The receiver incorporates a hybrid feature that permits the use of gaseous fuel when insolation is unavailable for longer periods than can be accommodated by the buffer storage. The design of the GE receiver has been completed (ref. 13), and portions have been fabricated, but the receiver has not been assembled or tested. This receiver can supply 30 to 45 min of storage time, depending on the operating mode.

For space application (low earth orbit), storage times exceeding 0.5 hr will be required (fig. 13, adapted from ref. 14). In early design studies for solar dynamic systems in space, lithium hydride was the most frequently selected thermal storage material (e.g., refs. 15 and 16). However, in more recent studies (ref. 14) lithium fluoride seems to be the choice. Accounting for shade time in low earth orbit, it will be necessary to provide concentrators having roughly double the size required to operate the heat engine. That is, half the concentrator area will be dedicated to charging the thermal storage unit so that the heat engine can continue producing electric power during the shade portion of the orbit.

FIELD OPERATIONS AND TEST EVALUATION METHODS

The experience gained in the terrestrial program will contribute significantly to future design, operation, and ground testing of space-developed components and subsystems. This experience includes installation, maintenance and repair, fault identification, and daily operation of autonomous modules. Further aspects include concentrator bore sighting, mirror alignment, mirror reflectance monitoring, mirror washing and care, receiver mounting and alignment, materials and insulation development for safe sun acquisition and emergency de-track, controls troubleshooting, power conversion installation and test, and data acquisition and analysis.

The two tools that have been used to characterize concentrator performance are the cold water calorimeter and the flux mapper (ref. 6 and 17). The calorimeter is essentially a receiver operated at near ambient temperature conditions to minimize thermal losses; it measures total thermal power, and thus optical efficiency of the concentrator, as a function of aperture size. The flux mapper can determine solar flux distributions in planes perpendicular to the optical axis of the concentrator, at the focal plane and at positions in front of and behind the focal plane. For a given calorimeter aperture size, the integrated flux distributions within the circular aperture should correspond to the total collected thermal power as measured by the calorimeter.

Indeed, early tests and data analysis indicated that excellent agreement could be obtained between calorimeter and flux mapper data provided that proper experimental precautions were realized. In additions, intercept factors determined experimentally were found to be in excellent agreement with theoretical predictions for concentrator performance. As noted previously, receiver performance and theoretical predictions were more disparate, because of deficiencies in both theory and experiment (e.g., uncertainties in wind conditions and cloud cover interruptions). It turns out that precise measurement of direct and indirect insolation has an important influence on all data correlations.

A variety of tests were performed to determine suitable insulation materials for protecting focal plane equipment and apertures (ref. 6 and 18). Techniques were devised to permit routine, safe sun acquisition as well as to protect against potentially dangerous sun walk-off (drive failure causing the solar image to proceed, i.e., "walk", across the receiver aperture plate).

Much experience was gained in data acquisition and reduction. A variety of computer programs were developed to facilitate data manipulation and display. Because of the many interacting factors, numerous parameters to be measured, and variable environmental conditions, data analysis for a complete module is a complex undertaking.

RELEVANCY TO SPACE POWER APPLICATIONS

Whereas most of the hardware developed in the terrestrial program is not specifically applicable to space flight conditions, the experience gained in design, fabrication, and testing is, nevertheless, relevant. A variety of optical reflecting surfaces have been developed and tested. It has been demonstrated that concentrators can be operated autonomously and efficiently over long periods of time. A

wide range of receiver designs utilizing five different working fluids have, in general, been operated successfully at temperatures ranging from 400°C to 1400°C. Many effective controls schemes have been devised and tested for subsystems and systems. Mechanisms have been explored that provide safe, routine sun acquisition and emergency de-tracking.

Designing hardware for space application is, in some ways, both easier and more difficult than for terrestrial application. Concentrators, for example, need not sustain gravitational, wind, seismic, hail or snow loads in space, and are not subject to blown sand, dirt, and acid rain. However, they must be packagable into small volumes and sustain launch acceleration. The environmental conditions of space are, of course, greatly different than on earth.

Receivers should operate much more efficiently in space. The only heat losses are by radiation (aperture, outer shell, struts, etc.), because there is no free or forced convection. However, because terrestrial work has focused mainly on short-term buffer storage, a great deal of work will be required in the area of long-duration thermal storage.

Large deployable reflector (LDR) concepts should be explored in depth for possible application to solar dynamic systems. For some years, NASA and JPL have been engaged in developing LDR concepts for communications and experimental astronomy purposes. In reference 19, for example, fourteen different concepts have been surveyed with respect to unique features, level of maturity, and development status.

If ground testing of future receiver, storage, and heat engine subsystems is to be pursued by NASA for space applications, it is recommended that a versatile concentrator, such as the TBC discussed herein, be employed. The TBC design offers great flexibility because it can be operated over a wide range of thermal power levels, and is amenable to custom tailoring of the focal plane solar flux distribution by means of symmetric mirror adjustment.

TECHNOLOGY BARRIERS CRITICAL TO SPACE

Although there is a 25 year history of solar dynamic technology development for space, most of that technology is obsolete. Very early work was done on Stirling systems but most of the hardware development was on Rankine and Brayton systems, and none of that was flight qualified.

Until detailed technology assessments are undertaken, there will remain disagreement concerning the viability of solar thermal dynamic systems for space application. Listed below are some technology areas that will bear close investigation and assessment.

- Integrated receiver/thermal storage concepts
- Stability of reflecting surfaces (vacuum outgassing, UV, protons/electrons, free oxygen, temperature cycling, micrometeorites, etc.)
- Concentrator design (deployable/erectable, pointing/tracking, mounting, articulation, etc.)
- Concentrator/radiator packaging
- Thermal and power control management systems
- Heat engines (reliability, lifetime, replacement)

- Working fluids (liquids and phase-change materials in zero gravity).
- Radiator design concepts
- Performance/cost compared to alternative systems
- Overall safety, reliability
- Modularity, i.e., clustering for larger power levels

CONCLUDING REMARKS

This paper summarizes some of the experience gained on the JPL terrestrial solar thermal parabolic dish program in the areas of concentrators, receivers, and storage technology. Heat engine development, including Stirling, organic Rankine, and Brayton cycles, also a major part of this program, is not covered herein. Many critical technologies have been demonstrated successfully. Recently, the first completely integrated parabolic dish module developed by the Advanco Corp., and featuring an improved Stirling kinematic heat engine, set a new solar-to-electric conversion record of approximately 30%. This module is in operation on Southern California Edison land at Rancho Mirage, CA. The combined experience of DOE, JPL, and industry gained in the TPS program will be a valuable asset in future NASA endeavors towards solar dynamic systems for space power applications.

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Table 1. Summary Characteristics of Some Point-Focusing Concentrators

DESIGN ORIGINATOR	SITE	DEVELOPMENT STATUS	OPTICAL CONCENTRATION*	OPTICAL MATERIAL*	POWER SUPPORT STRUCTURE	HOIST*	DRIVE*	CONTROLS*	DIAMETER m	FOCAL RATIO	OPTICAL EFFICIENCY	GEOMETRIC CONCENTRATION RATIO	INTERCEPT FACTOR
Acurex	Edwards, CA	DD	ParP	Ag/Gl	Polyester-glass or cellular glass on ring truss.	EI pivot on frame. Frame rotates in az on concrete pier.	EI-DM	ST/PrG/Py	11	0.6	0.96	2000	0.99
Advanco	Rancho Mirage, CA	BIT	SphP	Ag/Gl	Sheet metal ribs on trusses.	Off-horizontal bearing on frame for el & az. Frame on off-vertical bearing for more az.	DM	ST/PrG/Py/PT	9.8	0.6	0.94	2200	0.98
Boring	—	PD	ParP	Al/PE	Steel sheet on ribs and rings, on truss.	EI bearings rotate in az on post.	DM	ST/PrG/Py/EGh	13	0.6	0.92	1800	0.98
E-System	Edwards, CA	BIT/B	SphP	Ag/Gl	Cellular glass on trusses.	EI transitions on trussed pedestals. Pedestals on central az pivot & 3 wheels on track.	EI-DM/Act Az/DM/PrG or PT	ST/PrG/Py/EGh	11	0.6	0.94	2600 (adjustable)	0.99
Ford	—	QCD	ParP	Ag/Gl	Glass on trusses, sunward of mirror.	Egl. 1 pt. on jackscrew on hour angle wheel, 2 pts. directly on wheel. Wheel on tripod trusswork.	Deci-DM/Scr HA/DM/Ch	ST/PrG/Py/EGh	12	0.4	0.85	1950	0.99
GE	Edwards, CA & others	Prot	ParP	Al/PE	Sandwich, wool core, polyester-glass face. Corrugated ribs sunward of mirror.	EI transitions to 2 pedestals. Pedestals on central pivot & wheels on rail.	DM/DM	ST/PrG/Py/EGh	12	0.5	0.70	2600	0.95
JPL	—	Qd	ParP	Ag/Gl	Ribs, transverse tube.	EI bearing on arm. Arm on az bearing on post.	EI-DM/Scr	ST/PrG/Py/EGh	8-11	0.6	0.95	2000	0.98

Status: BIT-Built, Qd-Concept, QCD-Conceptual Design, DD-Detail Design & Development, PD-Preliminary Design, Prot-Prototype.

Configuration: ParP-Paraboloidal Panels, forming paraboloid, SphP-Spherical Panels, forming paraboloid.

Material: Ag-Silver, Al-Aluminum, F-Film, Gl-Glass, PE-Polyester, 2-Second surface.

Mount & Drive: Act-Actuator, Az-Azimuth, Cbl-Cable, Ch-Chain, Deci-Decination, EI-Elevation, DM-Electric Motor(s), Egl-Equatorial Mount (others are az-el), HA-Hour Angle, Scr-Scraper, Wl-Wheel. All have 2-axis pointing.

Controls: Az-Azimuth, CC-Central Computer, Egh-Ephemerides, Mp-Microprocessor, P-Pointing, PrG-Pointing without visible sun, PT-Past tracking data, stored, ST-Sun Tracking, closed loop.

- NOTES: (1) Efficiency, concentration ratio, and intercept factor are estimates, either from references or by the author. Note development status. Since the sources differ and the basis for the estimates varies from conceptual estimates to test results, use extreme caution in comparing estimates for different designs.
(2) Design characteristics may change as development proceeds.
(3) For data source of each concentrator see reference 4.

Table 2. Design Characteristics of Concentrators Installed at PDTs

TBC CHARACTERISTICS

- Nominal 11 m diam. reflector
- Paraboloidal mounting structure
- 6.6 m focal length, $F/D = 0.6$
- 224 facets (center 4 are inoperative)
 - Second-surface silvered glass
 - 0.61 m x 0.96 m nominal size
 - 3 regions of different radii of curvature 13.2 m, 15.7 m, 16.1 m
 - Initial reflectivity 95% max
- Estimated slope error 1 mrad (0.057°)
- Tracking error, 1 mrad (0.057°)
- Concentration ratios up to 3500 (variable)
- Design weight at focus, 500 kg (can be modified to accept 1360 kg)
- Heat output up to 82 kWth at insolation of 1000 W/m²

PDC-1 CHARACTERISTICS

- 12 m diameter paraboloidal reflector
- Internal rib structure
- $F/D = 0.5$
- Linear reflecting surface, 12 gores
 - Initial reflectivity 78%
 - Min reflectivity 72%
- Estimated slope error 4 to 6 mrad
- Tracking error 0.125°
- Concentration ratio approx. 1500
- Design weight at focus, 680 kg max
- Heat output 75 to 80 kWth at insolation of 1000 W/m²

Table 3. Receiver Characteristics Summary

Engine Cycle	Rankine		Brayton		Stirling
Manufacturer	Ford	AiResearch	AiResearch	Sanders Assoc.	Fairchild
Working Fluid	toluene	steam	air	air	helium
Fluid Outlet Temperature, °C (°F)	400 (750)	705 ^a (1,300)	815 (1,500)	1,370 (2,500)	815 (1,500)
Aperture Diameter, cm (in.)	38 (15)	22.8 (9)	25.4 (10)	19.7 (7.75)	27.9 (11)
Integral Hybrid Design	no	no	no	no	yes
Efficiency (%) ^b	70 to 90	80 to 92	70 to 80	up to 90	85 (est.)
Maximum Pressure, MPa (psi)	5.5 (790)	14 (2,000)	0.25 (38)	0.7 (100)	14 (2,000)
Material	metal	metal	metal	ceramic	metal
Buffer Storage	yes ^c	no	no	yes	no

^aThis is the capability for the receiver.

^bTemperature-dependent.

^c135 kgm (300 lbm) of copper acts as integral buffer storage.

Table 4. Properties of Candidate Latent Heat Thermal Storage Materials

Salt with mole fractions	Melting Point, °C	Heat of Fusion, KWH/kg	Heat of Fusion, KWH/m ³	Cost, \$/kg
NaF	988	0.220	431	0.24
KF	856	0.130	254	0.97
LiF	848	0.289	526	5.80
.667 NaF+ .333 CaF ₂	810	0.163	364	0.22
.774 NaF+ .226 MgF ₂	830	0.181	396	0.29
.543 MgF ₂ + .457 CaF ₂	944	0.181	449	0.29
NaCl	801	0.135	208	0.07
Na ₂ CO ₃	858	0.077	152	0.44
K ₂ CO ₃	898	0.066	124	0.57

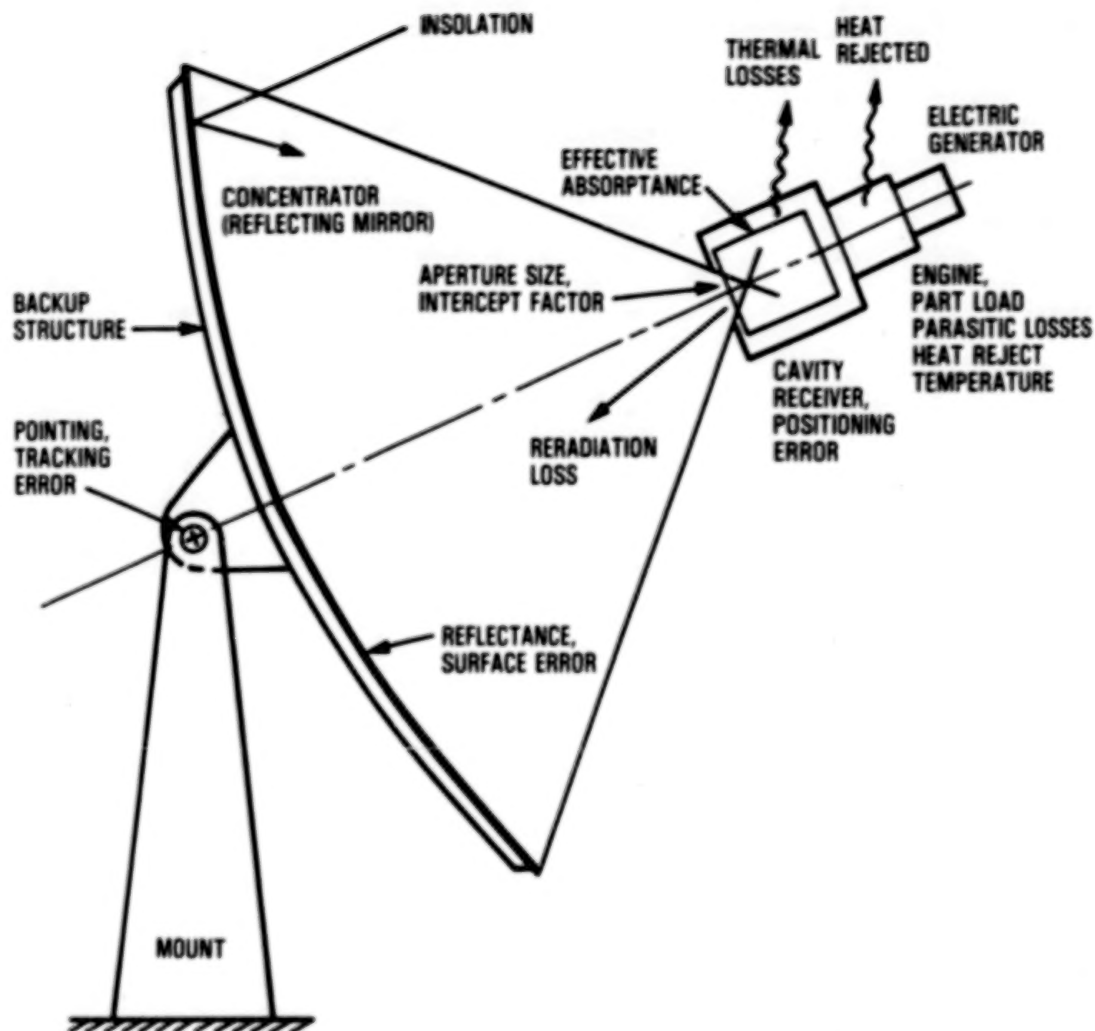


Figure 1. Schematic of Parabolic Dish Power Module, Major Subsystems and Losses

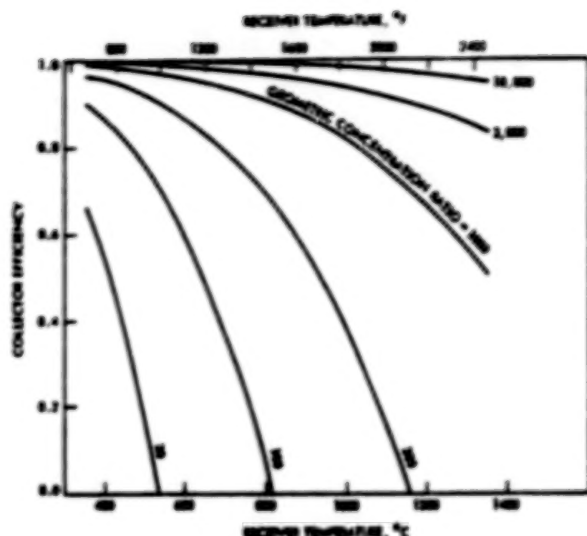


Figure 2. Effect of geometric concentration ratio and receiver temperature on collector and system efficiency. (Idealized system.)

Collector (concentrator plus receiver) efficiency. Insolation (solar flux) = 800 W/m^2 ; focal ratio = 0.6; optical efficiency = 1.0. Cavity receiver; absorptivity = emissivity = 1.0; receiver loss: only re-radiation through aperture.

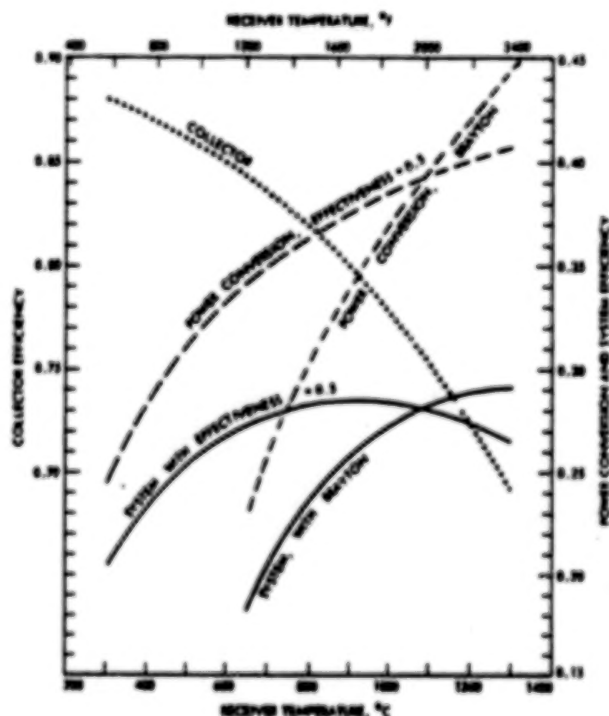


Figure 3. System efficiency vs. receiver temperature with engines of differing characteristics.

Effectiveness (percent of Carnot efficiency) of Brayton engines rises with inlet temperature; effectiveness of Rankine and Stirling engines is approximately independent of inlet temperature.

Insolation (solar flux) = 800 W/m^2 ; focal ratio = 0.6; reflectivity = 0.95; blocking and shadowing factor = 0.967; specularity = $0.5 \text{ mrad (} 1\sigma \text{)}$; slope error = $2.2 \text{ mrad (} 1\sigma \text{)}$. Geometric concentration ratio optimized at each temperature (receiver aperture adjusted) using Aparisi approximation for the flux distribution. Cavity receiver; effective absorptivity = 0.982; effective emissivity = 0.998. Receiver convective + conductive heat transfer coefficient = $73.6 \text{ W/}^\circ\text{C} \times \text{m}^2$ of aperture area. Temperature drop, receiver to engine = $20^\circ\text{C (} 36^\circ\text{F)}$. Brayton engine efficiencies per LeRC; efficiency of alternator + rectifier taken = 0.92. A fixed effectiveness may represent the variation of power conversion efficiency versus temperature for Rankine and Stirling engine; the numerical value of the effectiveness, however, depends on the particular engine; power conversion effectiveness = 0.5 is assumed here. Power processing efficiency = 0.93.

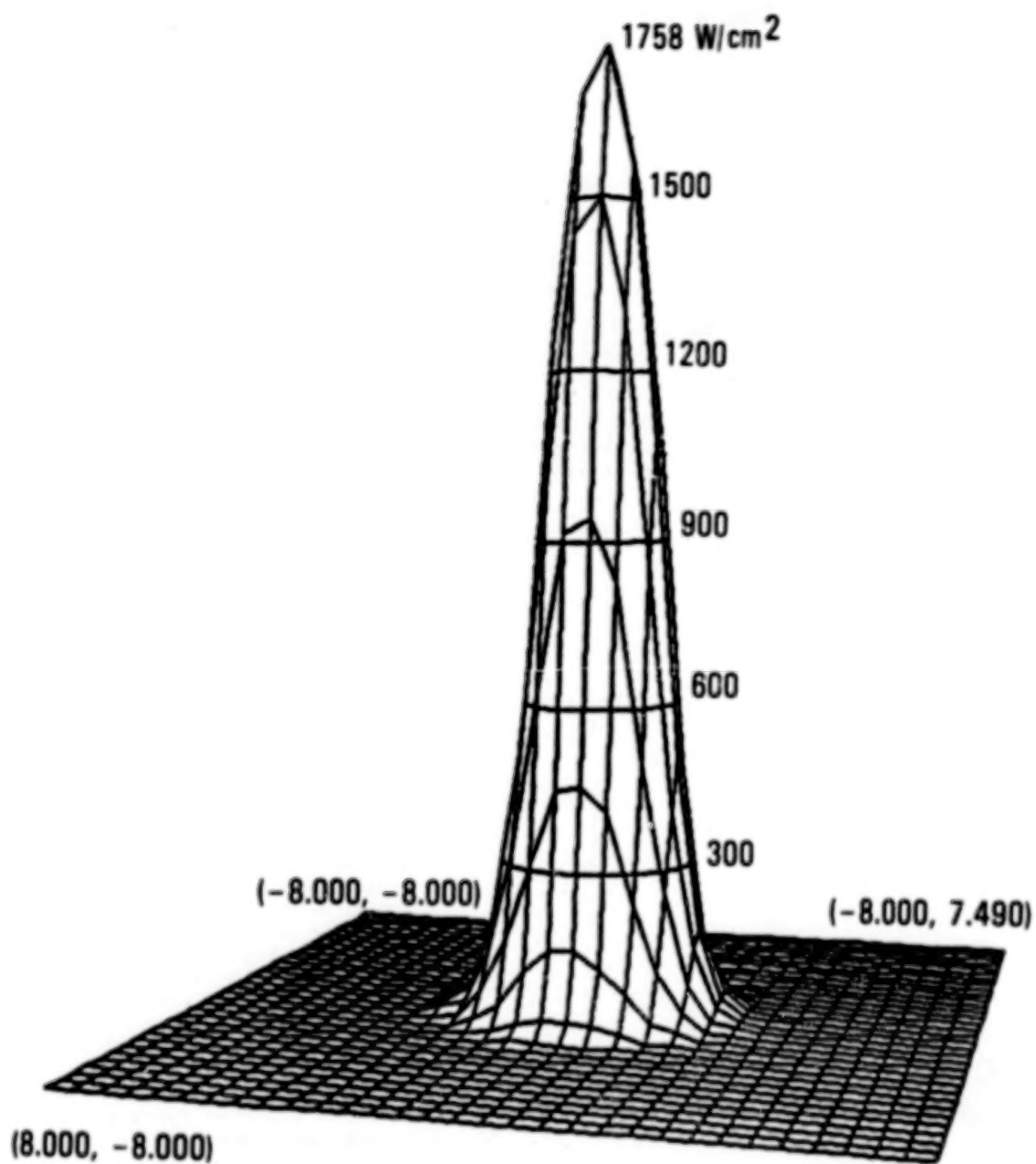


Figure 4. Focal Plane Flux Distribution, Test Bed Concentrator with 100% Mirrors

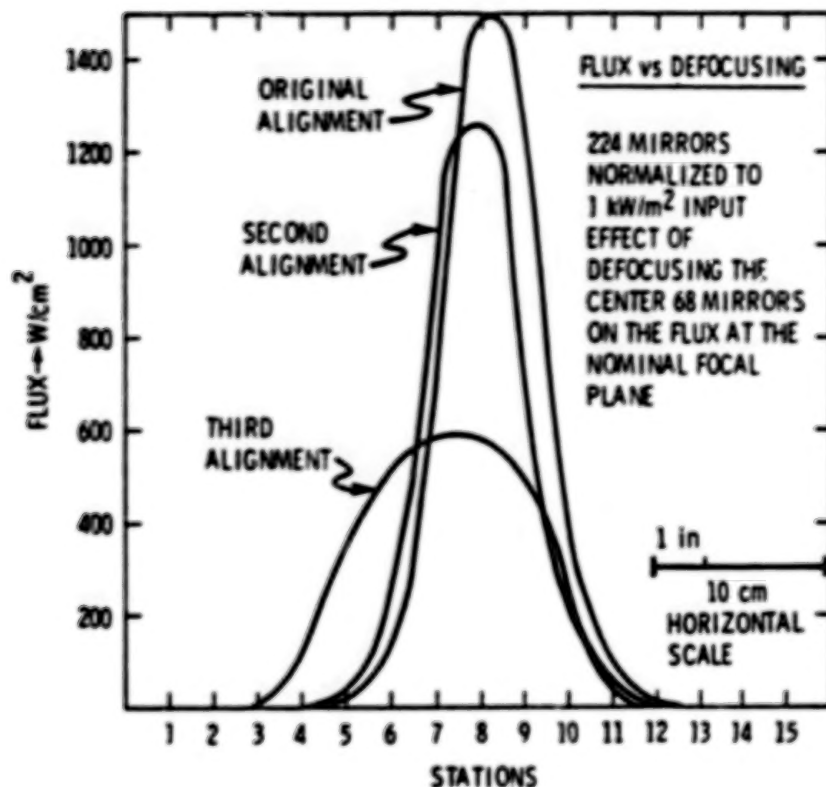


Figure 5. Solar Flux Measurements On Test Bed Concentrators

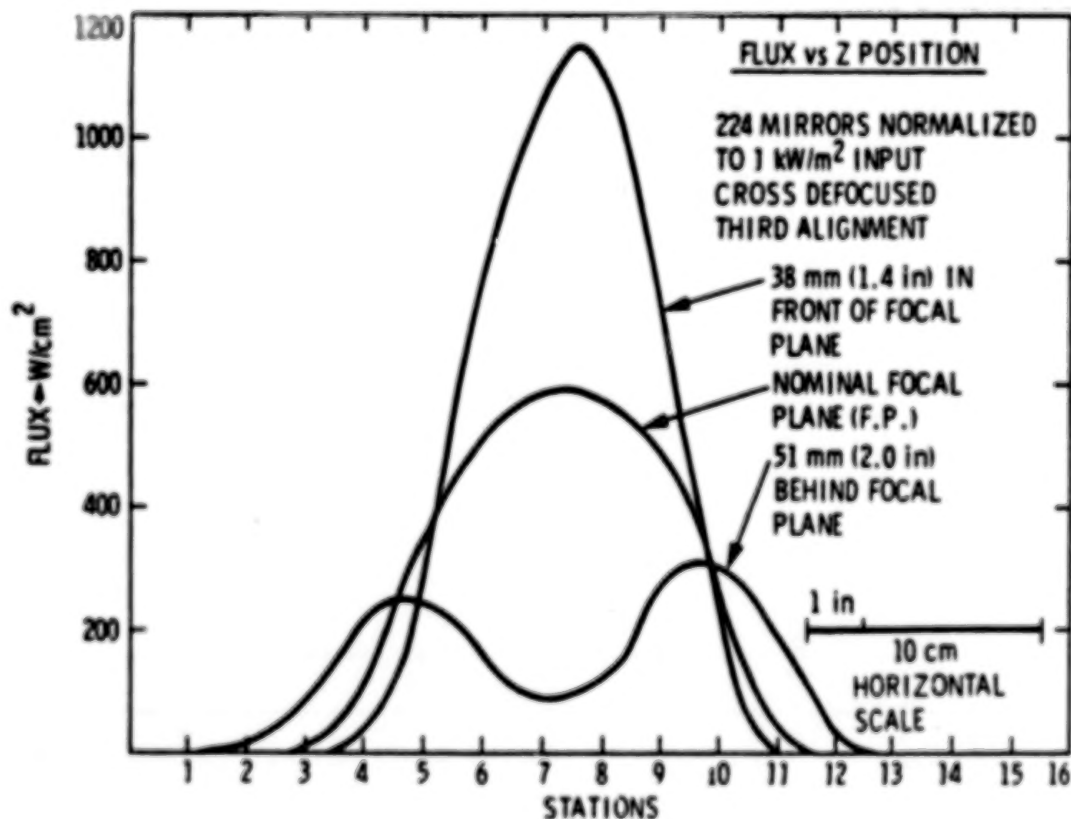


Figure 6. Solar Flux Measurements On Test Bed Concentrators

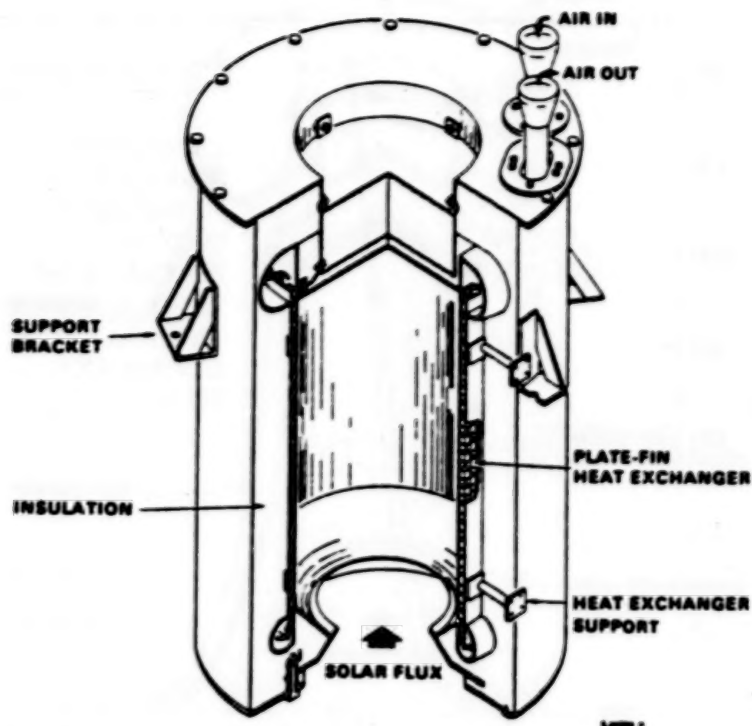


Figure 7. Prototype Air Brayton Receiver
(From ref. 9)

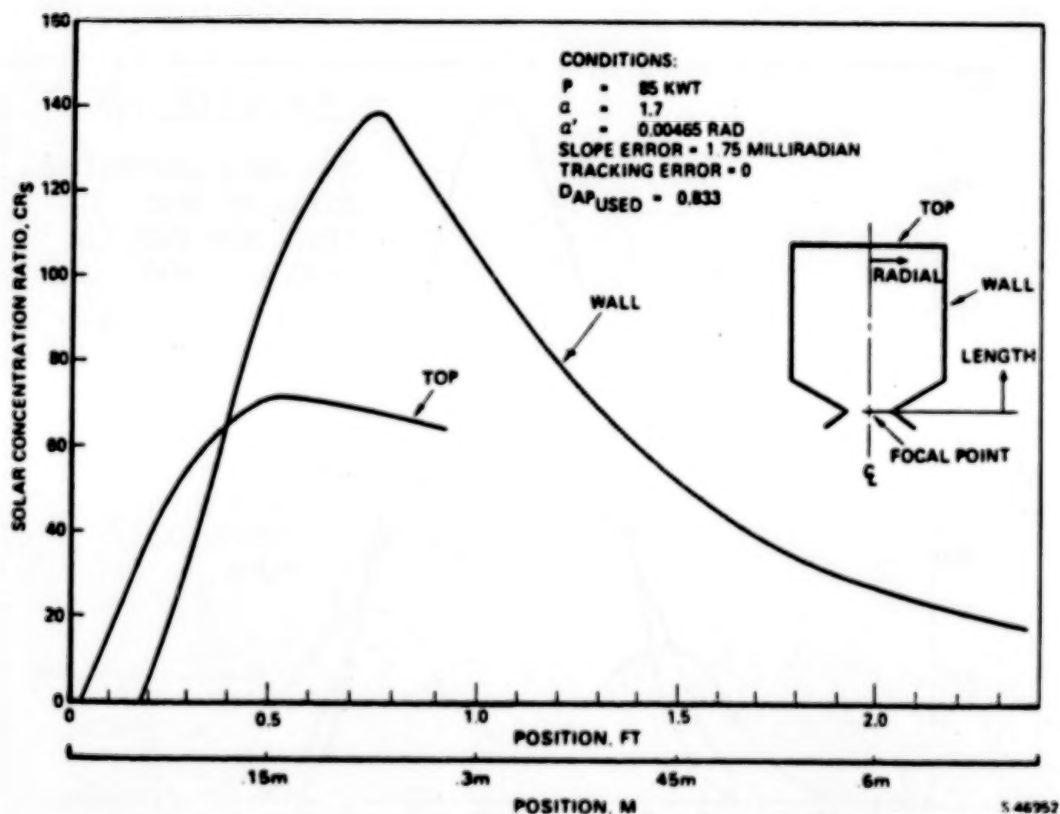


Figure 8. Cavity Wall Solar Concentration Ratio Distribution
(From ref. 9)

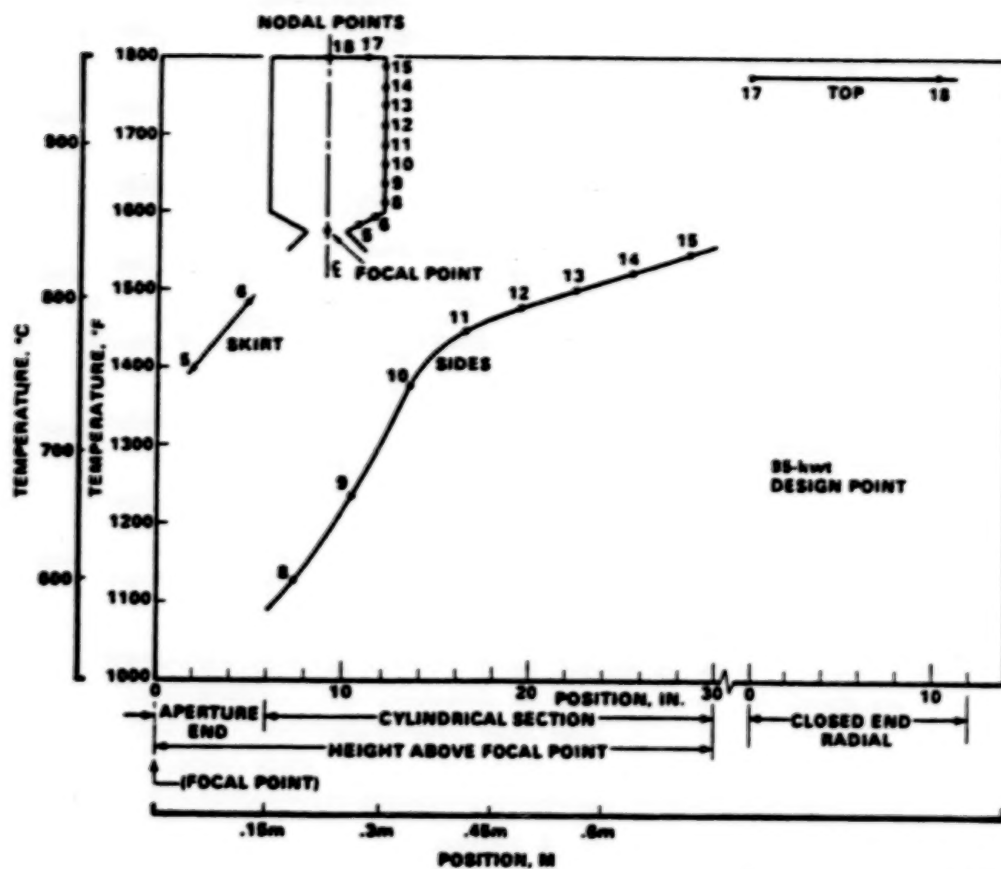


Figure 9. Steady-State Cavity Wall Temperature Distribution For 85-KWT Design Point Case (From ref. 9)

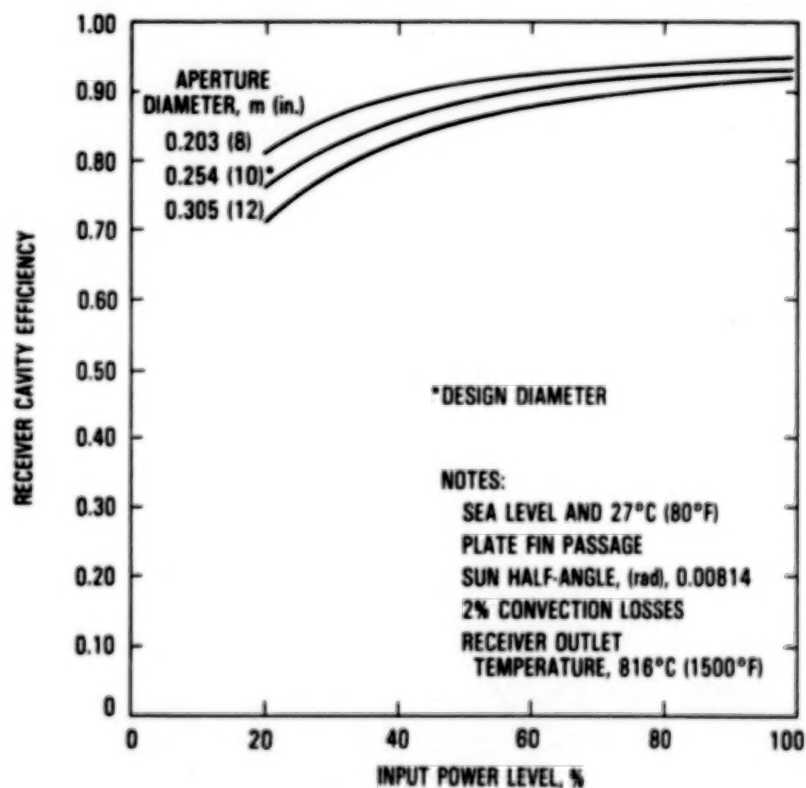


Figure 10. Brayton Solar Receiver Cavity Efficiency As A Function Of Power Level (From ref. 9)

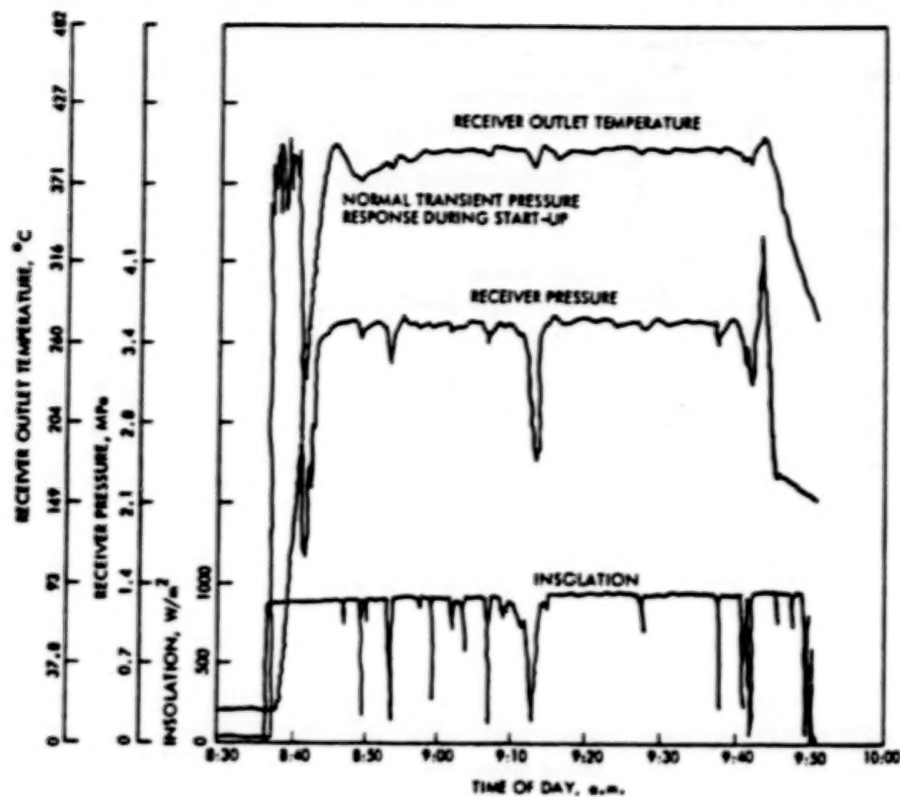


Figure 11. Effect Of Cloud Passage On Receiver Pressure And Fluid Outlet Temperature Of Ford Organic Rankine Receiver

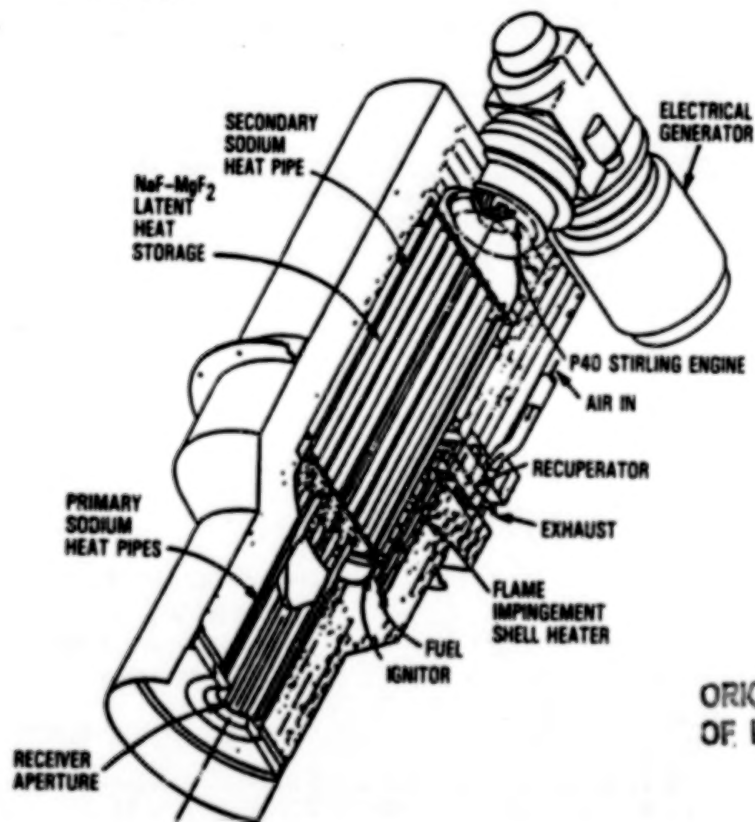


Figure 12. GE Dish-Stirling Heat Pipe Solar Receiver With Thermal Energy Storage

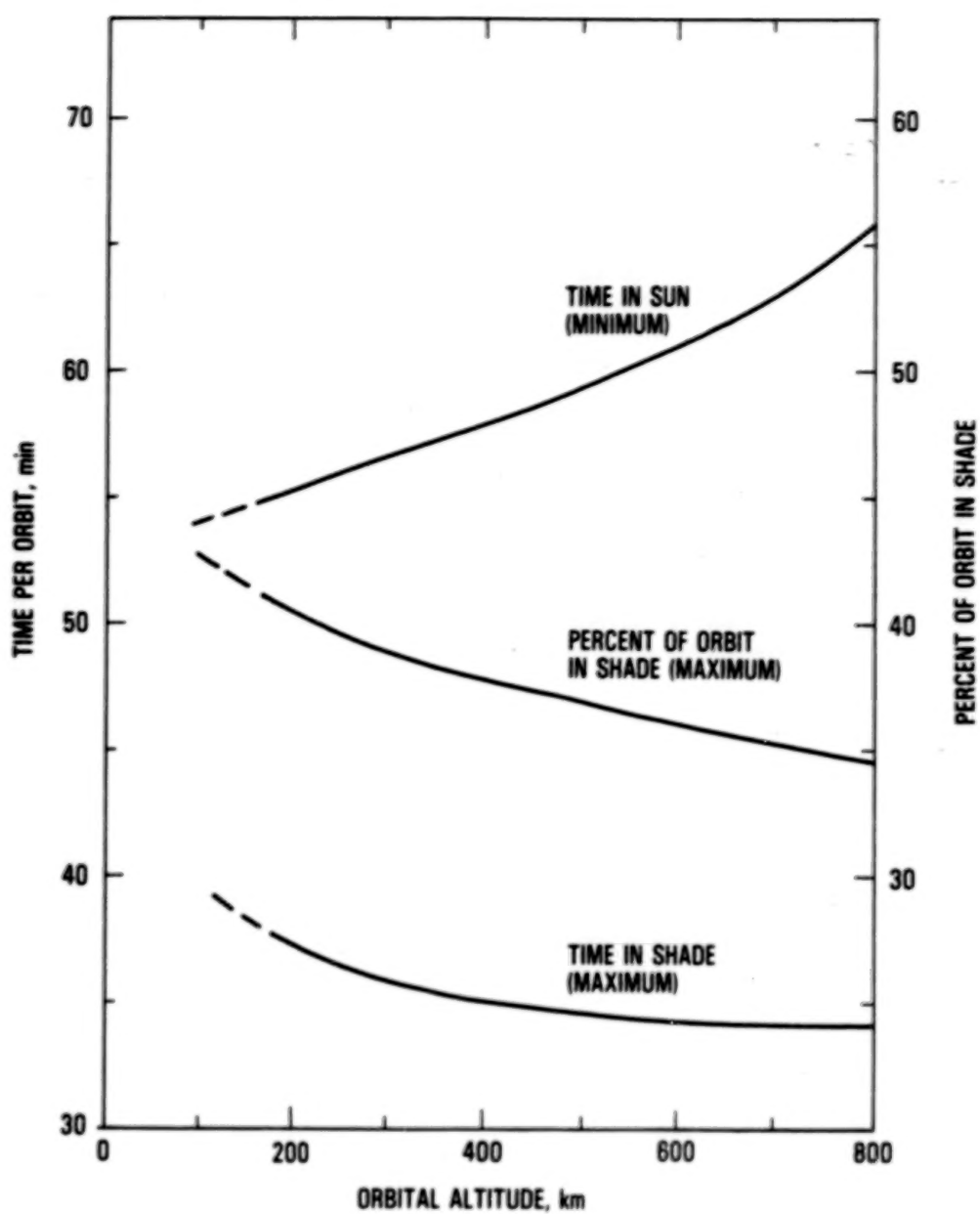


Figure 13. Sun And Shade Times As A Function Of Orbit Altitude
(Adapted From ref. 14)

DYNAMIC POWER SYSTEMS FOR POWER GENERATION

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Dynamic power-generating systems are suitable for use with either solar or nuclear heat sources. Apart from the heat sources themselves, the dynamic power-generating systems for these two heat sources are nearly identical, differing chiefly in particular adaptation or optimization for a given application. Thus, the technology required for use of either heat source is nearly the same.

Other characteristics of the dynamic power generators are advantageous in their integration with other subsystems. In addition to their compactness and ruggedness, their high efficiencies of power generation make especially effective use of a given energy source. If the Sun is the basic energy source, then this high efficiency reduces the size of solar collector required; in low Earth orbit, collector area might be only 1/3 - 1/4 that of a photovoltaic system producing the same steady power. If a nuclear reactor is the basic energy source, this high efficiency permits generation of more power from a given reactor or, alternatively, extending reactor life.

In some applications, these thermal power systems can also provide heat instead of some of the electric power required of photovoltaic arrays. For example, otherwise-wasted heat from the power generator can be supplied directly to an environmental-control and life-support system, energy demands that a solar array could meet only by providing electric power. For the Space Station, this might permit partial closure of the life-support loop at an early date, chiefly through distillation and purification of water from human waste. This heat can also reduce the need for electric power in a variety of other ways such as cooking food and heating water. Heat from the heat source itself might also be used in processing materials or as the heat input to the VM cryocoolers necessary for infrared sensors.

Thus, the characteristics of dynamic power systems have considerable potential value, especially for the Space Station. The purpose of this paper is to review the base of technology that makes these dynamic power systems practical. The following types of power-generating system are examined herein: organic Rankine cycle, potassium Rankine cycle, Brayton cycle and Stirling cycle.

BACKGROUND

One factor affecting the acceptability of dynamic power generators is the widespread misconception that they have inherently limited life and low reliability as a direct consequence of the motion of their components. In fact, one frequently hears the vacuous justification of alternate concepts on the basis of "nearly complete absence of moving parts." Just a moment of contemplation would dispel this myth.

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Consider a slide projector, for example. The projector's fan will almost surely outlast the lamp despite the fact that the fan and its motor rotate but the lamp stands still. The reason, of course, is that the lamp's filament operates at such high temperature, a condition leading to early failure of the tungsten filament. Although the tungsten is suitable for use at high temperature, even this material is pushed to extreme operating conditions in producing a very bright lamp. In contrast with this, the fan operates at fairly low speed and is built of materials readily tolerating the imposed stresses.

The question of life of a powerplant should thus be moved to more rational ground. Rather than merely asking if the powerplant has moving parts or not, one should inquire into both the severity of the imposed stresses (or operating conditions) and the ability of the powerplant's materials to tolerate these imposed stresses. The extent of the data forming the basis for such an evaluation should also be examined.

One set of data will serve as an example. For advanced dynamic power systems, a family of tantalum alloys was developed, T-111 (Ta-8W-2Hf) and ASTAR-811C (Ta-8W-1Re-0.7Hf-0.025C) being the most thoroughly investigated. Long-time creep tests (refs. 1-2) are summarized in the following table:

	T-111	ASTAR-811C
Number of tests	121	98
Total test time:hours	427 529	314 140
:years	48.8	35.8
Longest test:hours	38 129	23 694
:years	4.3	2.7
Temperature range, K	760- 1760	1140- 1920

In figure 1, the 1-percent-creep data for ASTAR-811C have been divided into ranges of high stress and low stress and plotted against Larson-Miller parameter LM, where

$$LM = T(15 + \log t)$$

where

T temperature, deg. Rankine

t time to 1 percent creep, hours

In each case, a straight line was drawn through the data by the method of least-squares, as in reference 3. The standard deviations of the data from these lines were also computed, and a second line drawn parallel to the first but shifted to lower stress by 2 standard deviations. For these 2-sigma lines, the stresses producing 1-percent creep in 40,000 hours were then calculated for various temperatures, as follows:

Temperature, K	Stress, MPa
1300	150
1400	97
1500	34
1600	7

The level of 34 MPa is appropriate for design of ducts, heat exchangers and turbine casings. Although higher strength is required for the turbine rotor, its temperature is lower, and materials of adequate strength are also available (ref. 3). Thus, materials are already developed for use in dynamic power systems at temperatures as high as 1500 K.

The conditions under which this judgment was drawn deserve some examination and emphasis. First, a substantial body of long-time data exists and has been correlated. Second, rather than relying on the correlated data themselves, scatter in the data was acknowledged by retreating by 2 standard deviations to lower stress. Third, the criterion for acceptable damage to the material (1% creep) is more conservative than failure (rupture). In fact, the expected creep of roughly 2.2 percent in 10 years is less than 10 percent of the deformation to produce rupture of this alloy in short periods of time.

In this manner, conservative use of a sufficient data base can provide a rational basis for both estimating and justifying life projections for dynamic power systems. On the other hand, dynamic power systems have a much lower multiplicity of power-generating elements than power systems based on, say, solar cells. Although redundancy can be readily incorporated into dynamic power systems, the number of redundant elements will be low in comparison with that of solar cells. This aspect occasionally leads both arm-wavers and computerphiles to claim inherently low reliability for dynamic power systems. Hopefully, comparisons of system reliability might be shifted to a more rational basis that stresses instead, as I have above, the extent of the data base, the allowance for variation among the data and the conservatism with which the correlated data are applied.

Figure 2 gives an overall perspective on several forms of power generation. In general, the dynamic power systems have efficiency potentials 3-5 times those of thermoelectric or thermionic power generation, these being alternate means for generating power from heat. Higher efficiencies are achievable with the Brayton cycle than with the Rankine, albeit at the cost of increased radiator area for a given peak temperature. The Stirling cycle, although at an early stage in its development, offers the promise of both high efficiency and low radiator area.

ORGANIC RANKINE CYCLE

Figure 3 is a representative schematic diagram of a Rankine-cycle powerplant. In that diagram, a pumped reactor coolant supplies heat to a boiler. Vapor passing through the turbine is condensed and the pumped back into the boiler. A third pump circulates a coolant that transports the heat from condensation to a waste-heat radiator. Power from the turbogenerator drives the pumps and operates the powerplant's controls, the remaining power being available to the useful loads.

The following table (ref.4) lists four fluids, candidates all for working fluids in organic-Rankine cycles, along with their temperature limits for thermal stability.

FLUID	STABILITY LIMIT, K
Dowtherm A	640
Fluorinol-85	560
Pyridine	640
Toluene	750

These temperature limits are at once a strength and a weakness, the limits being so low that there is no question about the strength and temperature-tolerance of the containing materials. Ordinary stainless steels can be used, thereby assuring easy fabrication and assembly. On the other hand, these fluids are incapable of exploiting the higher temperatures readily achievable with superalloys, much less the refractory alloys such as ASTAR-811C discussed earlier. These fluids can also lubricate any bearings in the power-generating system.

These temperature limits, although only approximate, must be adhered to with an almost religious fervor, for thermal decomposition produces the "noncondensable" gases hydrogen and methane. These products of decomposition would be transported by the working fluid to the condenser. The liquid-vapor phase separation occurring there would leave behind these noncondensable gases, adding their partial pressure to that of the vapor being condensed. If such gas evolution continues, the condenser will eventually be choked by the gases, and power output will decline severely.

An alternative is to periodically purge the condenser of these noncondensable gases by opening a valve in order to vent these gases to space. This would, of course, entail the loss of some working fluid along with the noncondensable gases, but this would be tolerable if the decomposition is small.

Such a vent valve is a site for a single-point failure in the system. That is, if the valve either fails to open or fails to close when it should, the powerplant would die. Addition of such a vent valve would increase the questions concerning reliability and durability of dynamic power systems.

Depending on powerplant design, turbine-inlet temperature might be limited to values below this thermal-stability limit. In boiler design, the thermal-stability limit must be imposed on the boiler's hotspot, generally a region of low flow velocity in a corner or in an otherwise inconsequential gap or crack. On the other hand, sufficient foresight and caution in boiler design could minimize this problem.

The organic-Rankine concept for producing power in space has been investigated off and on for about 25 years, the most recent effort being the Dynamic Isotope Power System (DIPS). The performance achieved is summarized in the following table:

Fluid:	Dowtherm A
Peak temperature:	630 K
Endurance test:	500 hours
Efficiency goal:	0.18
Efficiency achieved:	0.16
Power level:	1.3 kWe
Radiator area:	5.6 m ² /kWe (175W/m ²)

BRAYTON CYCLE

Figure 4 is a schematic diagram for the Brayton cycle. For use in space, the working fluid is generally a mixture of helium and xenon, the proportions being selected to produce the desired rotor-tip speeds for the compressor and turbine.

This blend of the lightest and heaviest inert gases has a higher thermal conductivity than would a pure gas (such as argon) of the same average molecular weight, a factor improving performance of the powerplant's heat exchangers.

Government-supported R&D continually advances the technology of gas turbines for various applications, and an energetic industry thrives on the design, manufacture and sale of gas turbines. Figure 5 illustrates the gas-turbine research conducted at NASA-Lewis, this compressor requiring 10 megawatts to drive it and having an efficiency of 0.905; the person in the photo shows the approximate size. The compressor in figure 6 is 6 meters in diameter and requires 100 megawatts to drive it; again note the person in the photo. The efficiency of this compressor is 0.91. These photos illustrate that design and manufacture of gas turbines in the power range of 1 - 100 megawatts are commonplace.

In the space-power program, our goal was to extend the existing technology and the available industrial experience to much lower powers. In particular, we investigated how far power output might be cut while still maintaining good overall efficiency. This initially led to investigation of compressors and turbines like those in figure 7, the radial-flow compressor having a rotor diameter of 152 mm (6 inches) and the axial-flow compressor a diameter of 89 mm (3.5 inches). Again note the person in the photo. The success of this early work encouraged us to explore even smaller turbomachinery (fig. 8), the smaller diameters being 81 and 89 mm for the compressor and turbine, respectively. The effect of these size reductions on efficiency of the compressors and turbines is shown by figure 9.

The design of a gas-turbine powerplant was based on this technology, rated power output being 10 kWe. Figure 10 illustrates the concept for the Brayton rotating unit (BRU), and figure 11 shows the actual machine. This BRU incorporates a radial-flow compressor, a radial-flow turbine and a synchronous alternator, all on a common shaft. Three gas bearings use the cycle's working fluid to support this rotor, two being journal bearings and one a bi-directional thrust bearing. Crucial to the concept's durability is that, during operation, the rotor does not contact the stationary parts. And use of the working fluid as the bearing lubricant avoids any contamination of the gas loop by oil. Inasmuch as the alternator itself is inside the gas loop, no shaft need penetrate the powerplant envelope, thereby avoiding any possibility of loss of the working gas through a shaft seal.

The complete powerplant (except for its heat source) was assembled, and its performance tested in the large vacuum chamber of the Space Power Facility (fig. 12). The power-generating system itself and the electric heater (ref. 5) simulating the actual heat source are at the bottom of the photo. The white cylinder above is the waste-heat radiator coated with heat-emitting but sun-reflecting paint. A cylindrical coldwall cooled by liquid nitrogen and a bank of lamps simulated the thermal environment of space.

Performance testing continued for 2561 hours, measured performance being given in figure 13. For the rated power output of 10 kWe, overall efficiency was 0.29. This efficiency is based on the net electric power available to the users after deduction of all power consumed by the powerplant itself for such things as controls, generator excitation, power conditioning and a motor-driven pump. Additional endurance testing continued in air to a total of 38,000 hours.

Some of the powerplant's components were tested individually, modified in the light of these tests and retested. For example, resetting by 3 degrees the vanes at the

compressor exit raised compressor efficiency by 0.04, and more slowly diverging the turbine-exit duct raised turbine efficiency by 0.01. Modifications to the control system decreased its power consumption by 400 watts, and recuperator effectiveness was raised from 0.94 to 0.95. Although these improvements were all demonstrated at the component level, they were not retrofitted into the powerplant. Had they been, its overall efficiency would have risen from 0.29 to 0.32 (ref. 6). The measured and predicted performances are summarized in the following table:

Peak cycle temperature: 1140 K
Power output: 10 kWe
Endurance test: 38,000 hours
Demonstrated efficiency: 0.29
Radiator area: $4.4 \text{ m}^2/\text{kWe}$ (225 W/m^2)

Improved components tested: compressor, turbine, electrical subsystem
Efficiency computed for substitution of these components: 0.32
Corresponding radiator area: $3.8 \text{ m}^2/\text{kWe}$ (260 W/m^2)

These efficiencies were all at the low power of only 10 kWe. Future demands for power will require modules of higher power output, perhaps at least 25 or even 50 kWe per module. Such a shift moves the Brayton concept closer to the powers used every day in aircraft propulsion and in power generation by central stations here on Earth.

Use of the tantalum alloy ASTAR-811C discussed earlier would permit raising turbine-inlet temperature from 1140 to 1500 K, the major effect being a 10-fold reduction in radiator area, as shown by the following table:

Peak cycle temperature: 1500 K
Predicted efficiency: 0.25
Predicted radiator area: $0.35 \text{ m}^2/\text{kWe}$ (2800 W/m^2)
No test of prototypic system

POTASSIUM RANKINE CYCLE

The potassium-Rankine powerplant would utilize the same family of tantalum alloys in order to boil potassium at about 1365 K and to superheat it to 1400 K at the turbine inlet. While employing the same materials as the 1500-K Brayton concept just discussed, the Rankine is limited to turbine-inlet temperature slightly lower because of higher temperatures imposed on the first stages of the turbine rotor. Although a great deal of component technology has been evolved (ref. 3), no prototypic power-generating system has been built and tested. A system study (ref. 7) estimated the following performance:

Overall efficiency: 0.19
Radiator area: $0.15 \text{ m}^2/\text{kWe}$ (6500 W/m^2)

This concept thus has the potential for very high performance. At a given peak cycle temperature, the radiator required is only 1/3 that of the Brayton concept. The most likely application of potassium-Rankine is at powers above 500 kWe.

STIRLING CYCLE

The Stirling cycle is a concept for a reciprocating engine whose working fluid is a high-pressure gas, either hydrogen or helium. Historically, the N.V. Philips Company of the Netherlands evolved the basic, enabling technology for the modern Stirling engine over about a 40-year period (ref. 8). Although the Stirling engine has been investigated for space power, none of the programs has yet been fully successful (ref. 9).

The largest effort on Stirling engines has been sponsored by the Department of Energy (ref. 10), and this program provides the principal base of technology for the ongoing investigation of Stirling for space power. The major changes required for its application in space are summarized in the following table:

	Automotive	Space
Peak temperature, K	1100	1100
Rejection temperature, K	325	550
Life goal, hours	3500	90,000
Power-output device	shaft	generator
Engine type	kinematic	free piston
Lubricant	oil	gas
Piston-rod seal?	yes	no
Working fluid	hydrogen	helium

The current program on free-piston Stirling engines has several facets that altogether are pointed toward early demonstration of the necessary technology for use of Stirling engines in space at peak temperatures of about 1100 K. The joint Sunpower-Lewis effort on the RE-1000 free-piston engine has achieved the following performance:

Peak cycle temperature: 925 K
Sink temperature: 300 K
Measured efficiency: 0.33

This efficiency is very high, albeit with a very low sink temperature. Computerized analysis of this engine gave the following comparison:

Peak temperature: 875 K
Computed efficiency: 0.33
Measured efficiency: 0.29

At Mechanical Technology Inc. (MTI), a 2-kilowatt free-piston engine has demonstrated the following life:

Peak temperature: 975 K
Life already demonstrated: 1000 hours
Life goal: 10,000 hours

The Space Power Demonstrator Engine at MTI will produce 25 kWe, a size more appropriate to the applications envisaged. This free-piston engine is focussed on achieving high efficiency with a sink temperature half the peak temperature. The specific goals are as follows:

Power output: 25 kWe
Peak temperature: 650 K
Sink temperature: 325 K
Overall efficiency: 0.25

This low peak temperature permits use of readily available, cheap materials. Simultaneously, a materials program is investigating materials and fabrication methods to raise the cycle peak temperature to 1100 K in combination with the life goal of 90,000 hours.

If this program is fully successful, the Stirling engine has the potential for higher efficiency, lower mass and smaller radiator than the Brayton cycle, the cycle peak temperatures being taken equal for both concepts.

RECAPITULATION

In comparison with photovoltaic arrays as sources of power, the dynamic power systems share the features of compactness, ruggedness and high efficiency. These concepts can also use heat from either nuclear reactors or from the Sun. Thus each offers an evolutionary approach to power generation over the long haul. Their salient differences may be summarized as follows:

- (1) In its potential for space power, the organic Rankine cycle is severely limited by thermal instability of its working fluid.
- (2) The alkali-metal (potassium) Rankine cycle offers the potential for the smallest radiator, but its technology is difficult, requiring great care in manufacture and in testing.
- (3) Of the concepts for generating power from 10's of kWe to 10's of MWe, the Brayton technology is furthest advanced. Among these concepts for converting heat to power in space, Brayton has demonstrated both the longest life and the highest efficiency.
- (4) Although Stirling-cycle technology is still in the early stages, the concept offers the potential for smaller mass and smaller radiator area than Brayton. Alternatively, peak cycle temperature could be lowered.

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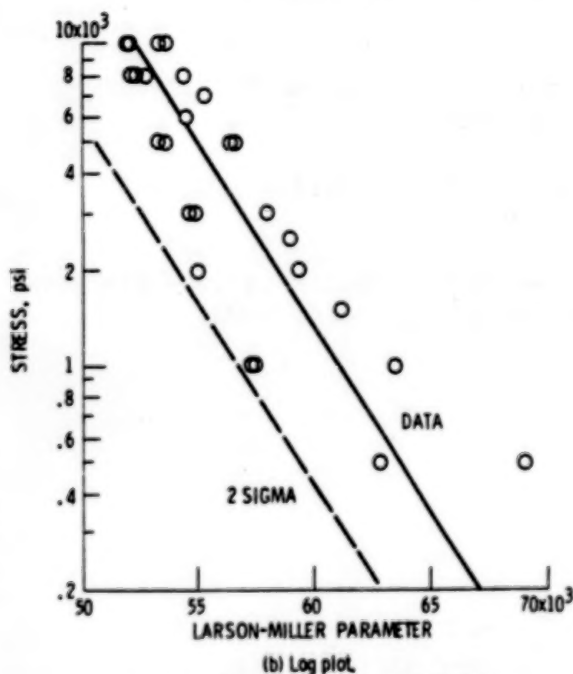
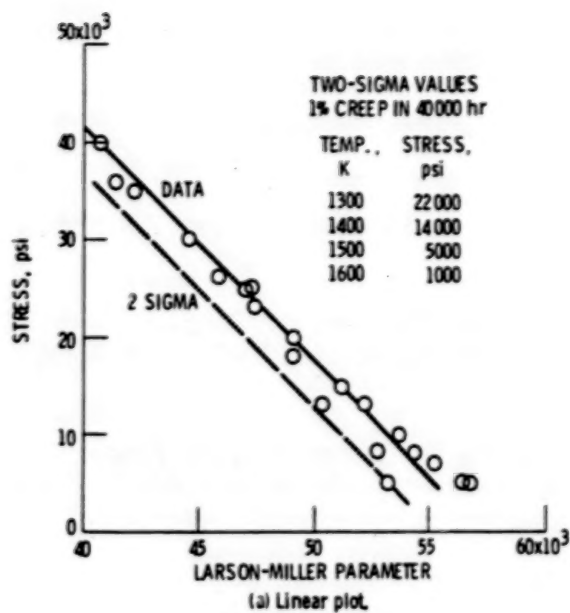


Figure 1. - One-percent creep of ASTAR-811 C.

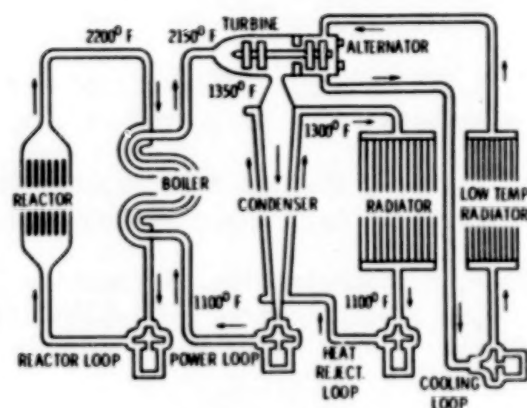
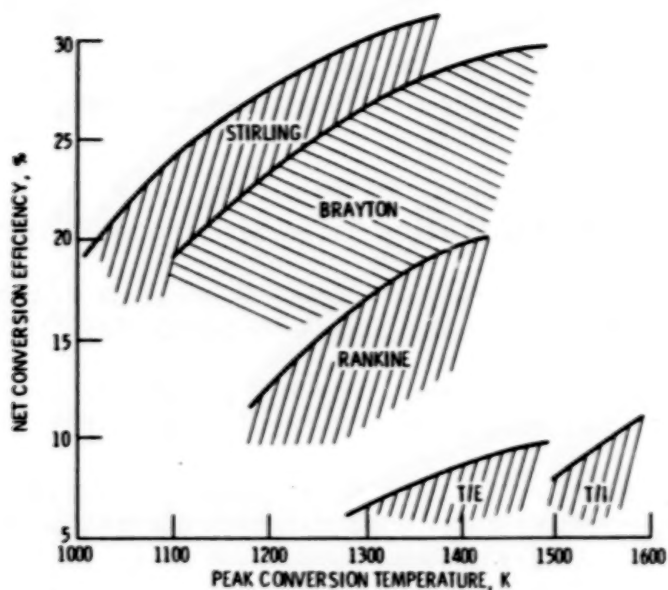


Figure 3. - Potassium - Rankine system.

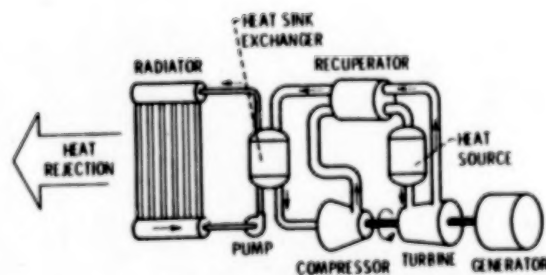


Figure 4. - Brayton-cycle space power system.



Figure 5. - Research compressor.

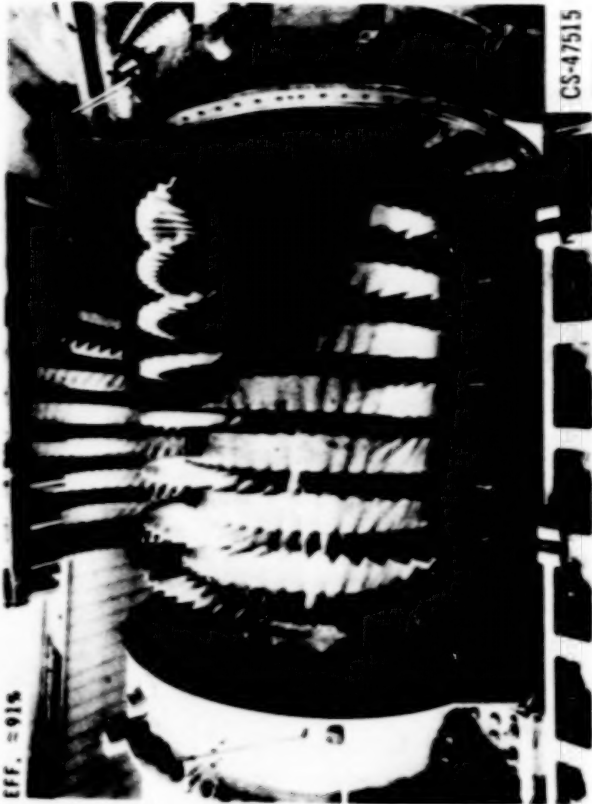


Figure 6. - 20 ft. diameter compressor.

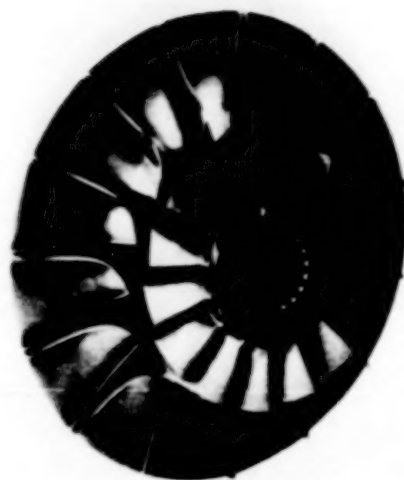
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(a) Centrifugal.
(b) Axial.

Figure 7. - Brayton-cycle compressors.

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(a) 6.0-inch diameter.



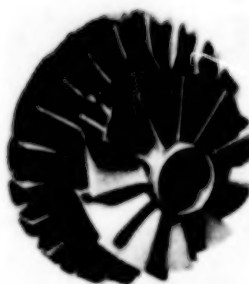
(b) 3.2-inch diameter.

CS-40268

Figure 8 (a). - Compressors for size effect study.



(a) 6.0-inch diameter.



(b) 3.2-inch diameter.

CS-40267

Figure 8 (b). - Turbines for size effect study.

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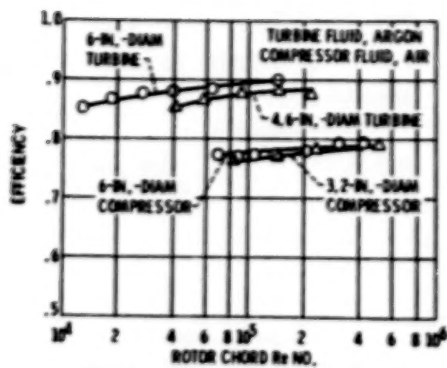


Figure 9. - Effects of Reynolds number and size on efficiency.

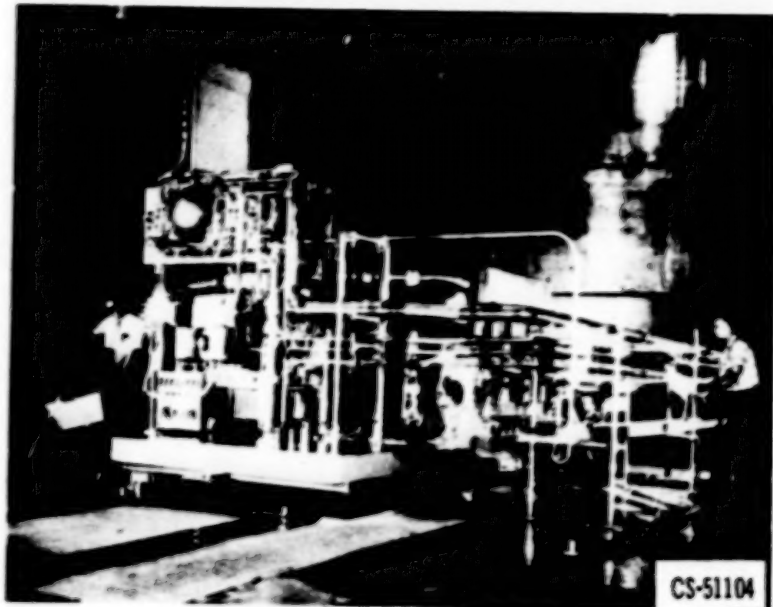


Figure 12. - Brayton system in SPF.

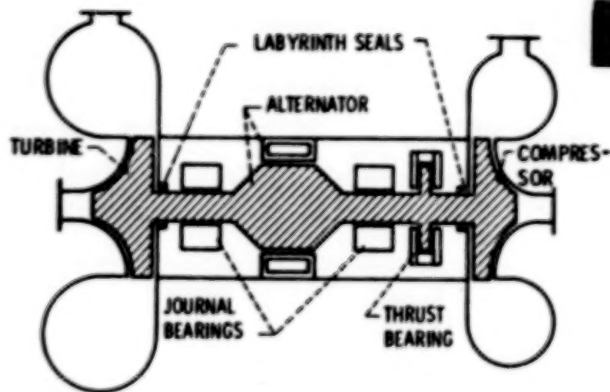


Figure 10. - BRU schematic.

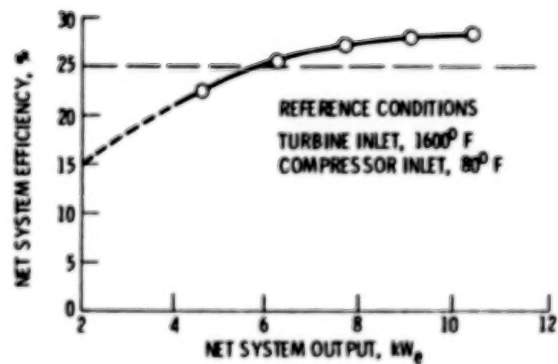


Figure 13. - Brayton engine performance.



Figure 11. - Brayton rotating unit.

TECHNOLOGY STATUS OF THERMIONIC FUEL ELEMENTS FOR SPACE NUCLEAR POWER

J. W. Holland and L. Yang
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During the intensive thermionic fuel element (TFE) development program from the mid 1960s to the early 1970s, over a half-million thermionic converter-hours of inpile and out-of-pile testing were accumulated in the US. When the program was terminated in early 1973, TFEs had operated 12,500 hours with projected 3-year lifetimes, and individual laboratory converters operated more than 5 years with stable performance. By the end of 1971 full-scale prototypical TFEs were being routinely manufactured and tested inpile. The high degree of quality yielded reproducible performances within ± 5 percent and no infant mortalities. Primary life limiting factors had been identified to be 1) thermionic emitter dimensional increases due to interactions with the fuel and 2) electrical insulator structural damage from fast neutrons. Multiple options for extending TFE lifetimes to 7-years or longer are available and will be investigated in the 1984 and 1985 SP-100 program for resolution of critical technology issues.

INTRODUCTION

Thermionic reactor power systems are particularly well suited for space missions requiring tens-of-kilowatts to megawatts of electrical power for several fundamental reasons:

- a) Thermionic reactor power systems have high optimum heat rejection temperatures, typically about 1000 K, which minimizes system mass and volume.
- b) Thermionic reactor power systems are static (no moving parts), and have many individual power producing elements. Both characteristics contribute to high reliability.
- c) The thermionic power conversion technology has room for growth in performance. As greater nuclear fuel capability develops, the high temperature capability of thermionic conversion can be more fully utilized.
- d) The technology is scalable over the kilowatt-to-megawatt electrical power range without new research and development.

This technology is currently being examined for the NASA-DOD-DOE SP-100 Project which requires 100 kWe for 7 years. In this application the thermionic reactor contains 172 thermionic fuel elements (TFE), each of which produces about 0.6 kWe. The TFEs, illustrated in fig. 1, contain 6 thermionic converters connected electrically in series. Figure 2 shows the components of the converters and the materials and functions of the components.

Development of thermionic converters started with single thermionic devices which were tested out-of-pile with electrical heaters. Some were cylindrical geometry converters constructed of the same materials that would be used in TFE construction. The Mark VI LC-9 is an example. Figure 3 shows the test results of LC-9 which operated at constant performance for over 5 years at an electrode power density of 8 watts/cm² and efficiency of 14 percent (ref. 1). The test was still fully operational at program termination in January 1973. We concluded from the results of this test and hundreds of thousands of hours accumulated in similar tests that there are no fundamental mechanisms in out-of-pile converters that would limit lifetimes to less than 7 years.

Over 320,000 hours of inpile converter testing had been accumulated by January 1973 (ref. 2). The majority of these hours were accumulated within TFEs. Figure 4 illustrates the development evolution from single cells to TFEs. As the development progressed, life limiting mechanisms were systematically identified and solutions found to reach the lifetime goal of 20,000 hours. The data base available in 1973 supported TFE lifetimes of about three years with emitter temperatures of 1800K. Primary life limiting factors were identified to be 1) thermionic emitter dimensional increases due to fuel-emitter interactions and 2) insulator structural damage from fast neutrons. The result of unrestricted emitter dimensional increases is shorting of the converter electrodes thereby reducing the output voltage to zero. Structural failure of an insulator seal will result in a reduction or loss of the cesium vapor which is necessary for acceptable performance. Structural failure of the sheath insulator can result in increased collector temperatures and off-optimum performance.

FUELED-EMITTER TECHNOLOGY STATUS

In the 1960-70 program, TFEs were fueled with either UO₂ or UCZrC. UO₂ was found to be superior in respect to performance stability and compatibility with the tungsten emitter (refs. 3, 4). Therefore, UO₂ was selected for the SP-100 baseline design. Figure 5 shows emitter distortion found in the UO₂ fueled TFE 6F3 as a function of operating time. Straight line extrapolation of the maximum emitter diametral increase as a function of time shows that the emitters operating at about 1780 K would have contacted the collector after about three years operation. Correlation of these data and other similar test data with viscoelastic analyses is shown on fig. 6. The dimensions shown are emitter thicknesses, with the existing data base covering the range from 1 to 2 mm. Thicknesses of most interest in the SP-100 program are toward the high end of the range. The correlation indicates that at 1700 K, emitter lifetime could be well above 4 years for 2 mm emitters. Further inpile testing and improved analysis are planned to confirm this prediction.

The tungsten emitters are fabricated in two steps, both using vapor deposition (ref. 4). The first step is the hydrogen reduction of WF_6 on a hot molybdenum mandrel. The result is a controlled fine grain columnar structure having superior structural qualities for this application. A second step involves vapor depositing a 0.3 mm thick second layer by the hydrogen reduction of tungsten chloride over the first layer. The deposition is controlled to provide a $\langle 110 \rangle$ preferred crystal orientation surface. This surface typically has a work function of 5.0 electron volts, the high value of which is favorable for thermionic conversion. Fast neutron irradiations of chemically vapor deposited tungsten commensurate with the SP-100 design requirements have not exhibited any significant deleterious changes to the material (ref. 5).

INSULATOR TECHNOLOGY STATUS

Lucalox alumina insulators irradiated in the TFE test program of the early 1970s were not subjected to sufficient fast neutron doses to affect their structural integrity. However, in a separate fast neutron irradiation program (ref. 6), insulator materials and assemblies were subjected to fast fluences sufficient to cause volumetric swelling such that the ceramic bodies were no longer leak tight, allowing the Cs and fission gases to mix. This occurred when the ceramic had swollen by about 3 volumetric percent. Figure 7 shows the band in which the alumina data fell. Data for very fine grained alumina occupied the right side of the band.

Fast fluences of interest for the future range up to 1.7×10^{22} n/cm² ($E > 0.1$ MeV). If fine grained alumina were used, the maximum useful life of the insulator would be about 3 years. Therefore, to meet emerging requirements, insulators with improved radiation resistance must be developed. Such development was initiated (ref. 7) in the 1970s using Y_2O_3 ceramic. In recent years the fusion program tested a number of alternate candidate insulator materials to very high fast neutron fluences. Materials that were demonstrated to have superior dimensional stability in addition to Y_2O_3 are $Y_3Al_5O_{12}$, Si_3N_4 and Si_2ON_2 (refs. 8, 9). The bands within which the volumetric expansion of these materials fall is shown in fig. 7. If the same 3 percent volumetric expansion criteria for leakage applies to these ceramics that applied to the alumina ceramic, the irradiation resistant ceramics should endure at least 7 years.

The requirement for leak tightness primarily applies to the insulator seal. In the earlier program the insulator seals, shown in fig. 8, were fabricated (ref. 7) by brazing tapered niobium cylinders to machined and metalized Lucalox ceramic bodies. The ceramic was metalized with tungsten-2 wt. % yttria. Seal brazing was performed in vacuum using vanadium-niobium alloy. The seals were qualified in out-of-pile testing operation at 1500 K.

The essential requirements for the sheath insulator assemblies are to electrically isolate the thermionic cells from the reactor coolant and to conduct the waste heat from the collector to the sheath which is in contact with liquid metal coolant. The sheath insulator assembly, sometimes called a trilayer, is a layer of ceramic bonded between the two layers of niobium. Niobium has been the material of choice because its thermal expansion coefficient closely matches that of alumina. To assure bond integrity throughout the required thermal cycles, the trilayers were fabricated by plasma

spraying a graded cermet over a niobium mandrel starting with 75-25 percent niobium/alumina and then progressing to 50-50, 25-75, 0-100, 25-75, 50-50, 75-25 ratios of niobium to alumina. A niobium sleeve was fit over the cermet and the assembly was gas pressure bonded to form a dense cermet structure. The resulting structure, shown in fig. 9, was used successfully in a number of inpile tests.

Ranken (ref. 10) irradiated trilayers in fast neutron fluences to 6×10^{21} n/cm² and found separation of the alumina ceramic from the inner niobium collector. The cause was postulated to be due to the fast neutron induced volumetric expansion differences between the niobium and alumina. In the same series of tests, a trilayer made with Y₂O₃ instead of Al₂O₃ did not show separation of the layers. Other candidate insulator materials found to be especially radiation resistant will be investigated in the resumption of trilayer fabrication development.

CELL AND TFE ASSEMBLY TECHNOLOGY STATUS

Operations in the manufacturing of TFEs are discussed in this section. The tungsten emitter is diffusion bonded to a tantalum cylinder by resistance heating the tungsten-tantalum joint (ref. 11). Figure 10 shows the location of the bond. Similar bonds have operated 10,000 hours inpile at prototypical temperature gradients without deterioration.

The remainder of the niobium-niobium and niobium-tantalum joints in the cells and TFEs are electron beam welded. All assembly operations were accomplished in a clean room facility, fig. 11, to which an electron beam welder and a high sensitivity leak detector are functional extensions. A schematic of a thermionic cell and a TFE in various stages of assembly are shown in figs. 12 and 12B. Pictures of finished components are shown in figs. 12C through 12I. Final processing involved bonding an outer sheath tube of Nb-1 Zr over the six trilayers, the final outgassing, and the loading of cesium into the TFE. The sheath tube bonding was accomplished by wrapping the sheath tube with tungsten wire over the trilayer areas and then heating to a 1500 K to cause the sheath tube to be compressively yielded by the differential thermal expansion of tungsten and niobium. A 2-3 micron thick layer of nickel plated onto the inside of the sheath tube before assembly assured an intimate bond. The bond was then inspected ultrasonically.

TFE manufacturing controls instituted in the earlier program yielded performance reproducibility within $\pm 5\%$, as shown in fig. 13, and zero infant mortality (ref. 3).

MULTIPLE OPTIONS FOR TFE LIFETIME EXTENSION

Alternative development paths are available for extending TFE lifetime beyond the 3 to 4 years found in the earlier program. Figure 14 identifies and classifies these multiple options for extension of TFE lifetime to well over 4 years. To resolve the fueled-emitter dimensional stability issue, design options, performance options and fuel material options are all available. Because emitter distortion has been found to be a strong function of emitter

temperature, one option is to lower the emitter temperature to 1700 K from the 1800-2000 K values used in the earlier program. Another design option is to increase the emitter thickness. The impact of these options is evident by inspection of fig. 6.

Increasing the interelectrode gap to allow greater emitter dimensional increases provides a third design option which is related to the performance options. High emitter work functions make TFE performance relatively insensitive to the interelectrode spacing. Figure 15 illustrates this insensitivity in a converter with a nominal $\langle 110 \rangle$ single crystal tungsten emitter (refs. 12, 13). The higher emitter work function reduces the required cesium pressure and the voltage losses and thereby allows larger spacings.

Several fuel material options to the baseline UO_2 fuel are being considered. Application of coated fuel particle technology similar to that developed under the HTGR program offers the potential of complete elimination of fuel clad mechanical interactions and fission gas venting. In this concept the fission gases, fig. 16, are retained within the ZrC coatings which serve as tiny pressure vessels (ref. 14). Another option is controlled porosity UO_2 which promotes release of fission gases and thereby reduces the pressure that can be applied by the fuel to the tungsten emitter.

For long life TFEs, irradiation resistant insulator materials will be used. As mentioned earlier, Y_2O_3 , $Y_3Al_5O_{12}$, Si_3N_4 and Si_2ON_2 have been demonstrated in the fusion program to be dimensionally stable in fast neutron fluences that far exceed the SP-100 requirements. With these ceramics growth to larger systems will not be limited by insulator damage effects.

TFE TESTING

A very important addition to the TFE development program in the late 1960s was the establishment of a dedicated thermionic fuel element test reactor (TIFR), shown in fig. 17. This TRIGA-type reactor used 70 percent enriched uranium. Erbium was added to the fuel to serve as a burnable poison and to enable core operating lifetimes of 9 megawatt-years. This feature provides for unperturbed TFE operation since the reactor was operated with very little control rod movement for extended durations and without the frequent refuelings normally experienced in fast reactors.

The reactor was designed for simultaneous operation of up to 15 TFEs. Although the fissions in the TFE are produced by thermal neutrons in TIFR, the fuel enrichment may be lowered in the TFEs to 5-10 percent to achieve a relatively flat radial power profile and a temperature profile nearly identical to that experienced in a fast reactor, thereby simulating the fuel irradiation conditions of a fast reactor. This is done with the TIFR operating at 3 MWt. At this power level the fast neutron fluence (>0.1 MeV) is the same as the peak fluence projected for space reactors, thereby exposing insulators to prototypical conditions. Another important capability of the TIFR facility is the unrestricted access for instrumentation and the capability for periodic neutron radiography. This latter feature allows for nondestructive measurements of the cell components as the testing progresses. This is particularly important to the determination of emitter deformation rates and the prediction of fueled emitter life.

CONCLUDING REMARKS

Our thermionic reactor development program plan calls for a state of TFE technology readiness consistent with a long (e.g. 7-year life, by October 1985). With that accomplished, thermionic reactor power systems with their low mass and volume radiators, high performance and multiple redundant energy conversion are a prime candidate for fulfilling space nuclear power requirements into the multimegawatt region.

The development of the thermionic reactor is strongly aided by several unique and inherent features of the thermionic fuel elements. First, the TFEs contain many of the essential features of the overall system-nuclear energy generation, energy transfer from the nuclear heat source to the energy conversion system, and thermionic energy conversion. Not only are all of these functions tested in individual TFEs but the functional interfaces are tested as well. Further, testing of TFEs in a test reactor has proven to be straightforward since each TFE produces only a small fraction of the reactor electrical output and multiple testing of TFEs is easily accommodated within a modest program. And most importantly, the high temperature portion of the thermionic reactor is limited to the fueled-emitter which is thermally isolated from the fuel element sheath tube and the remainder of the power system.

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SCHEMATIC OF IN-CORE THERMIONIC FUEL ELEMENT (TFE)



Figure 1.

TFE COMPONENT FUNCTIONS

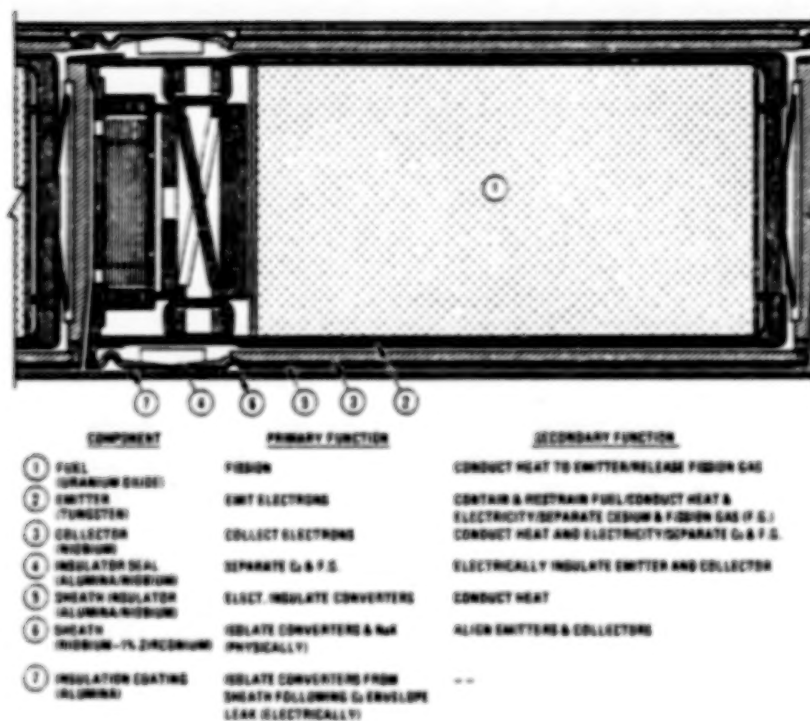


Figure 2.

OUTPUT STABILITY OF CONVERTERS

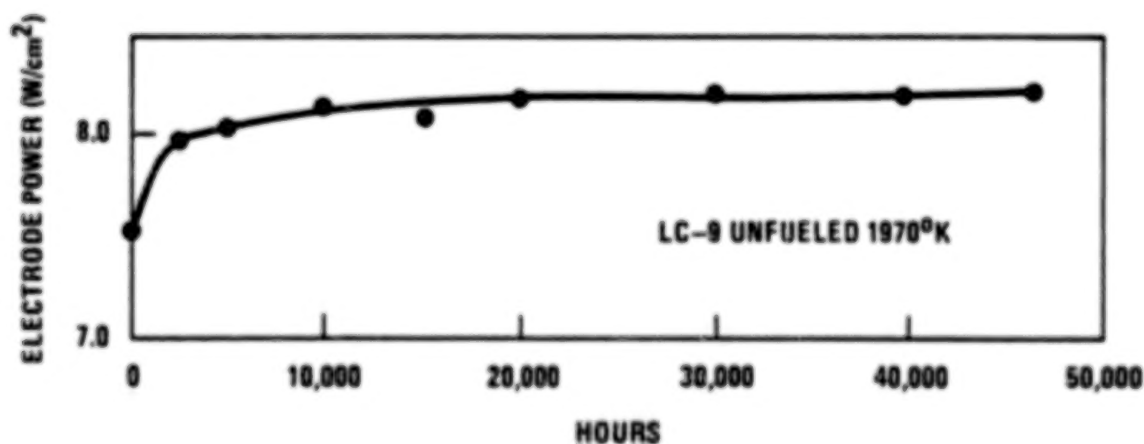


Figure 3.

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AEC-NASA FUNDED SUCCESSFUL TFE DEVELOPMENT PROGRAM

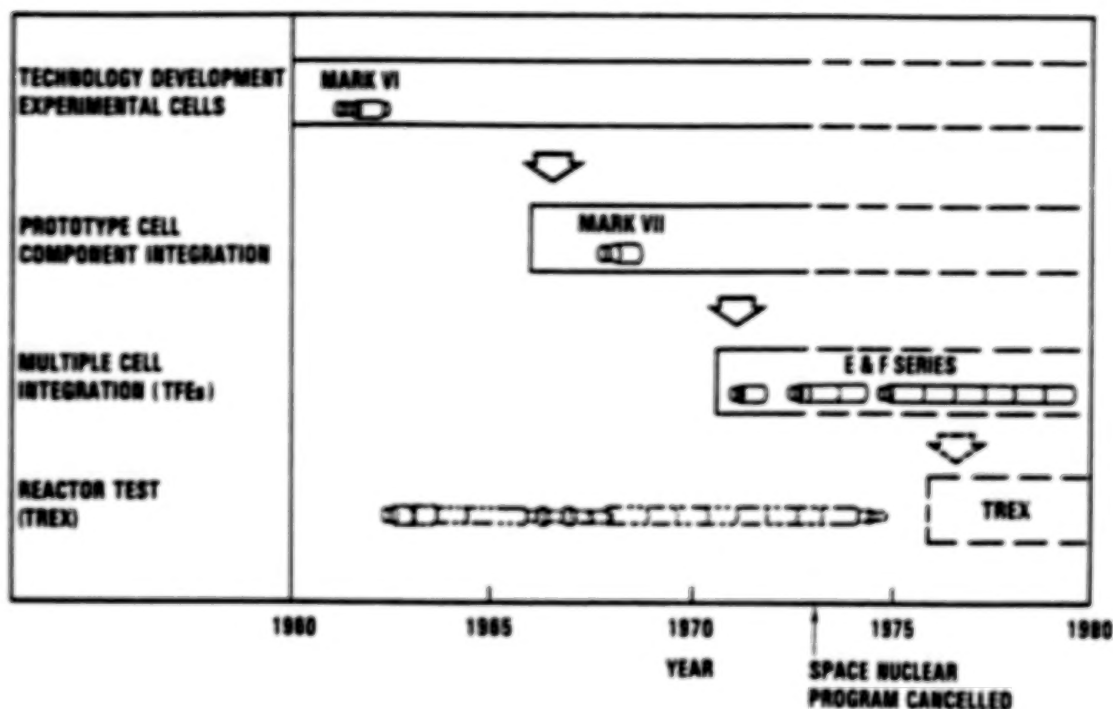


Figure 4.

TEST RESULTS ALLOWED FORECASTS OF TFE LIFETIME

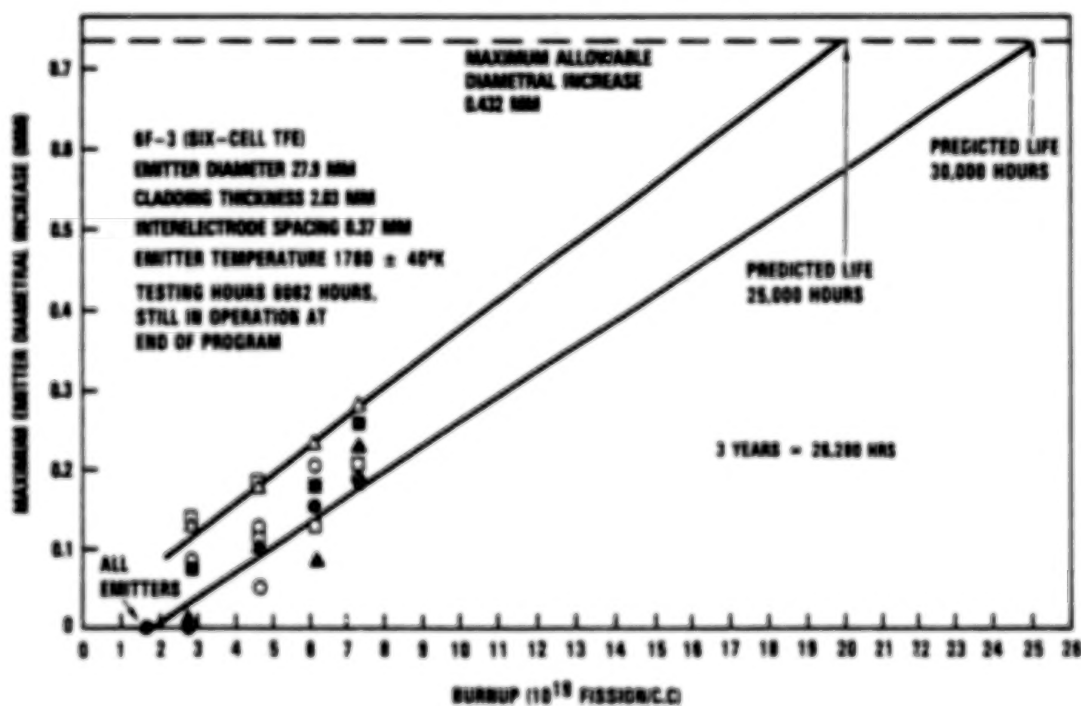


Figure 5.

TFE LIFETIME ANALYSIS

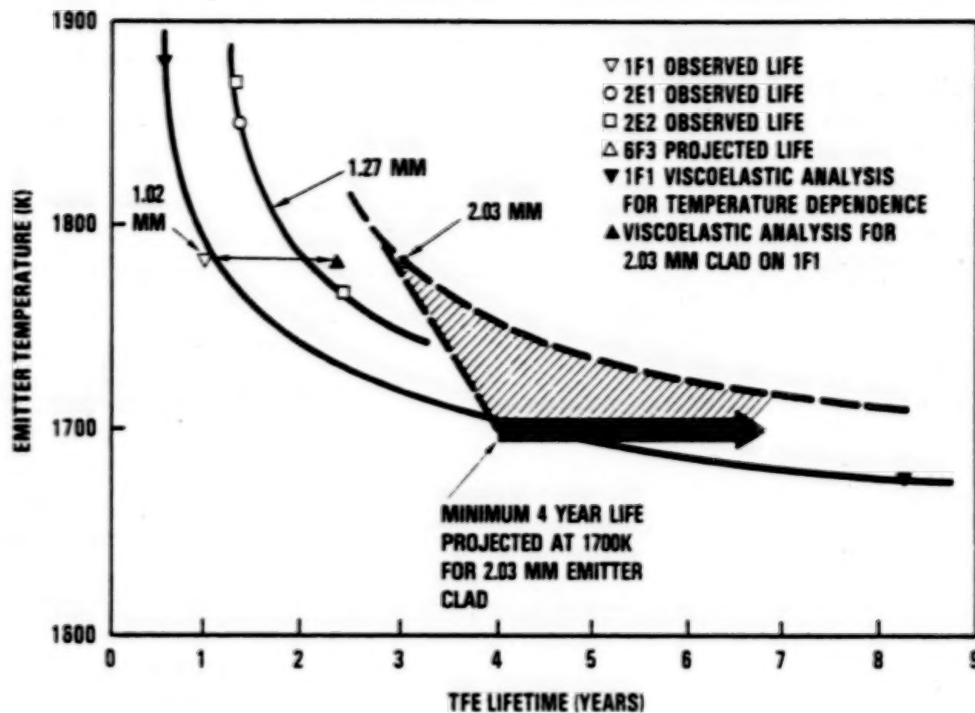


Figure 6.

INSULATOR LIFETIME IMPROVEMENT OPTIONS

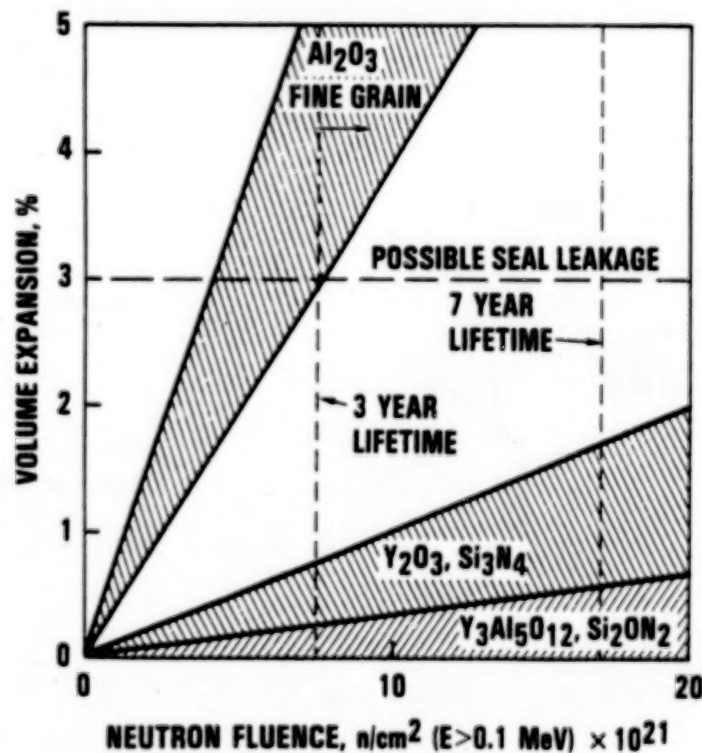


Figure 7.

INTERELECTRODE INSULATOR SEAL

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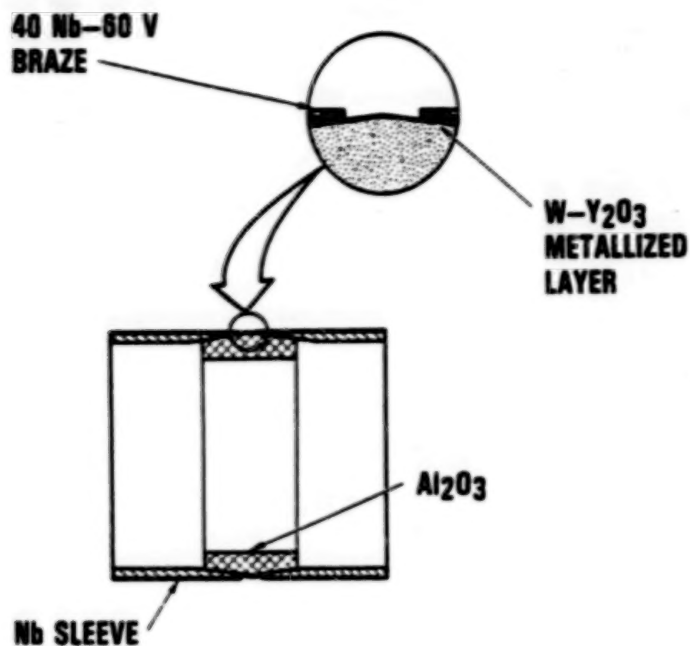


Figure 8.

TRILAYER CONFIGURATION AND MICROGRAPH

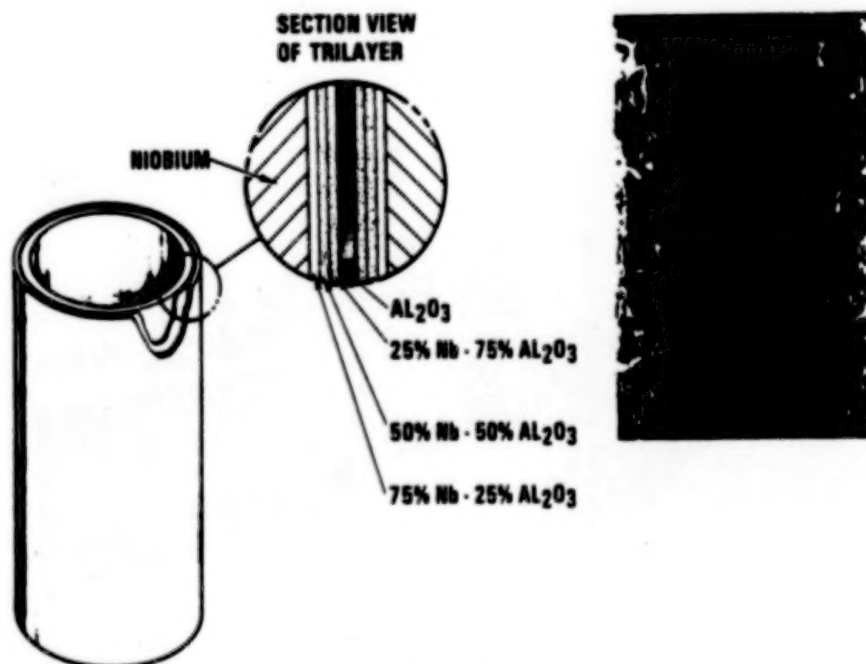


Figure 9.

LOCATION OF TUNGSTEN-TANTALUM DIFFUSION BOND

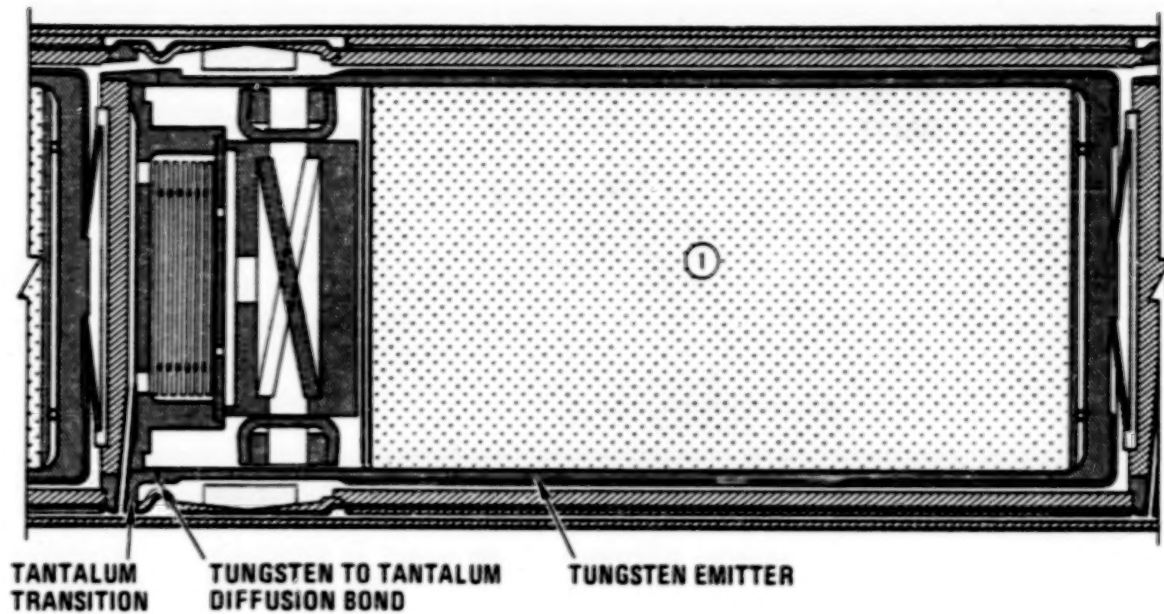


Figure 10.

QA OPERATION IN CLEAN ROOM



Figure 11.

F CELL FLOW CHART

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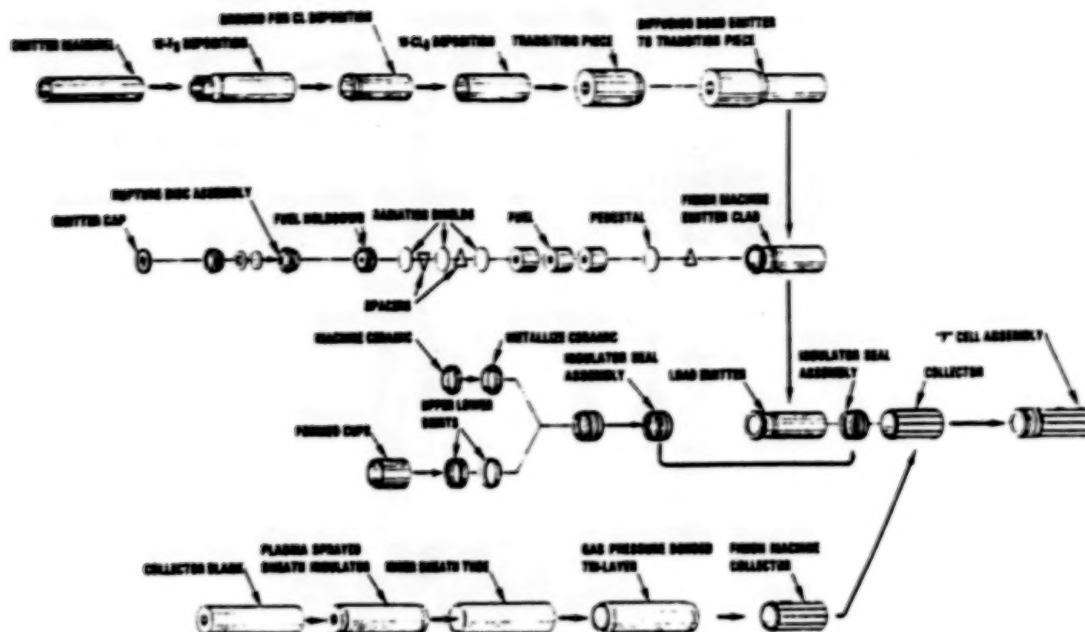


Figure 12A.

TFE-6F FLOW CHART

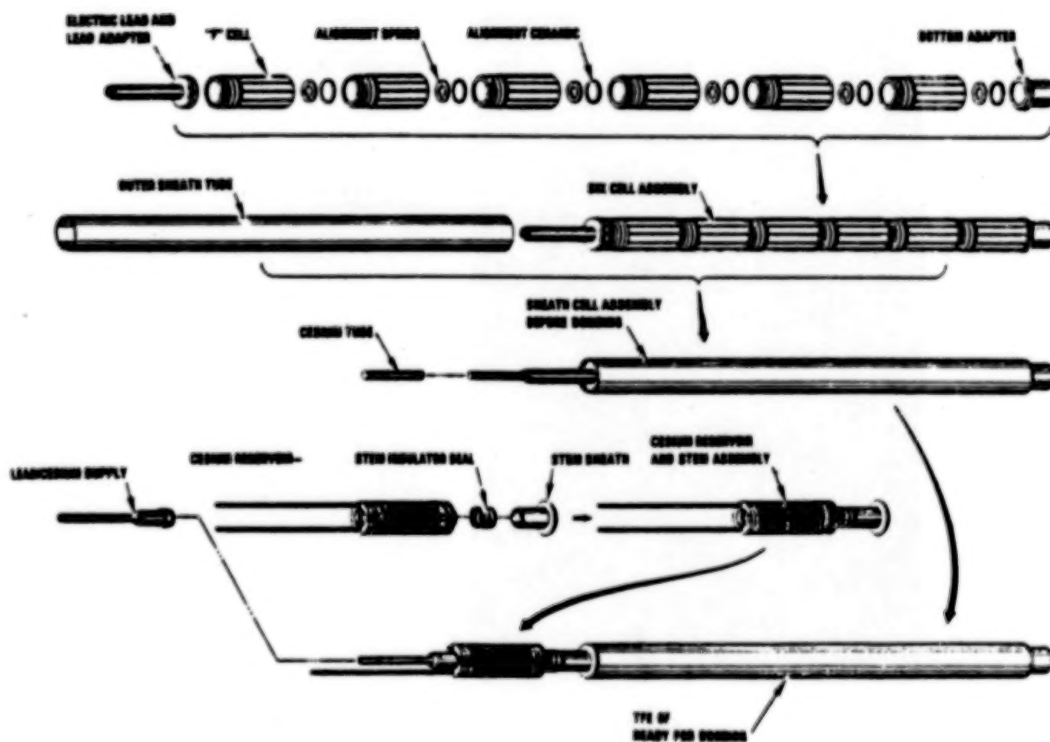


Figure 12B.

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TRILAYER, EMITTER AND CERAMIC-TO-METAL SEAL STRUCTURES FOR
F SERIES CELLS

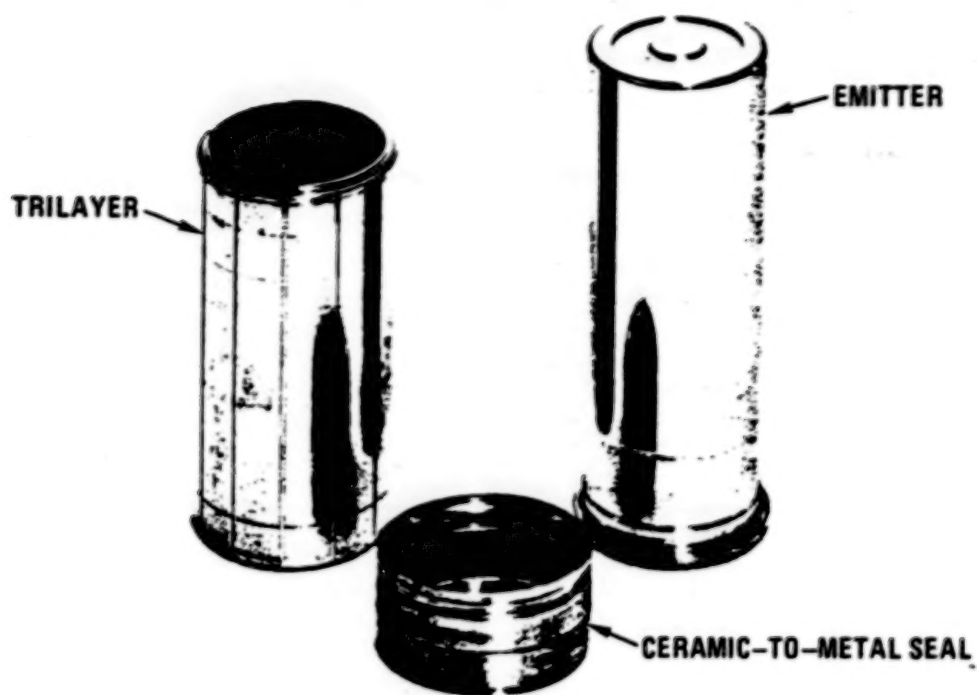


Figure 12C.

ASSEMBLED F SERIES CELL

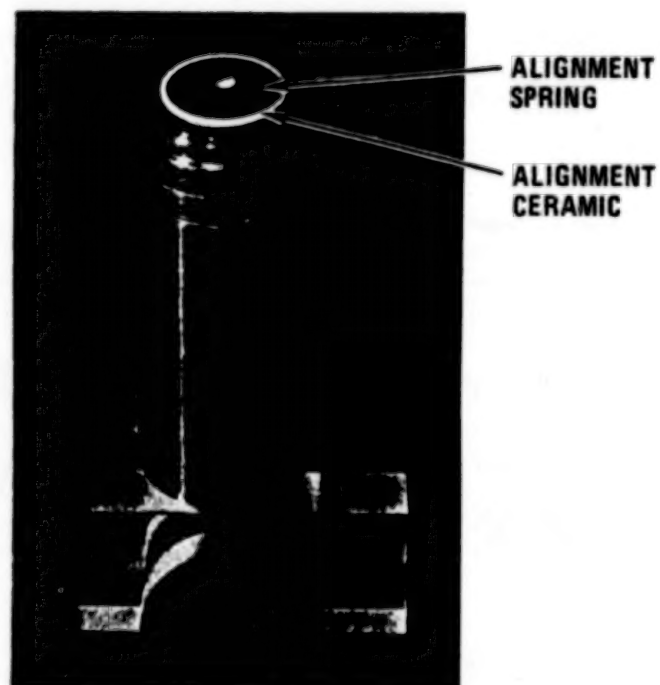


Figure 12D.

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PARTIALLY ASSEMBLED F SERIES TFE SHOWING ELECTRIC LEAD STRUCTURE

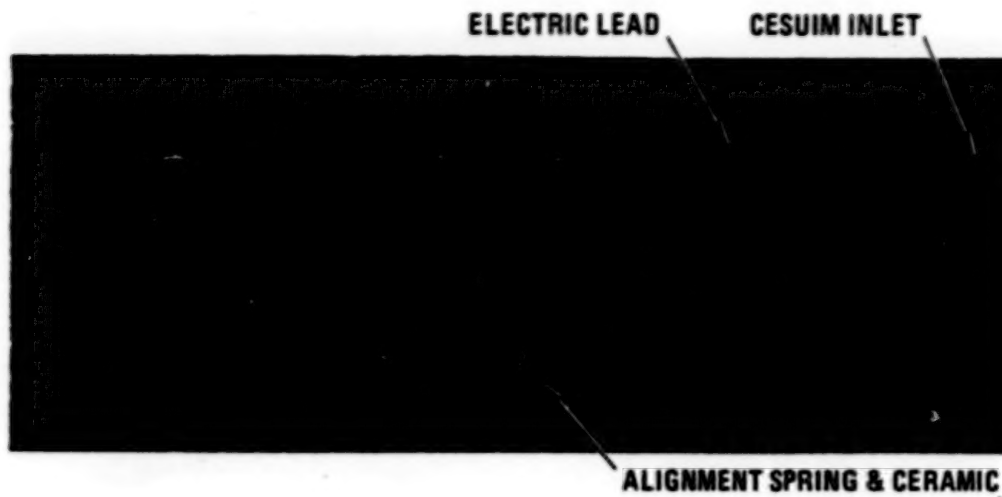


Figure 12E

6F TFE AFTER CELLS, ELECTRIC LEAD AND ALIGNMENT PIN HAVE BEEN ASSEMBLED
IN SERIES

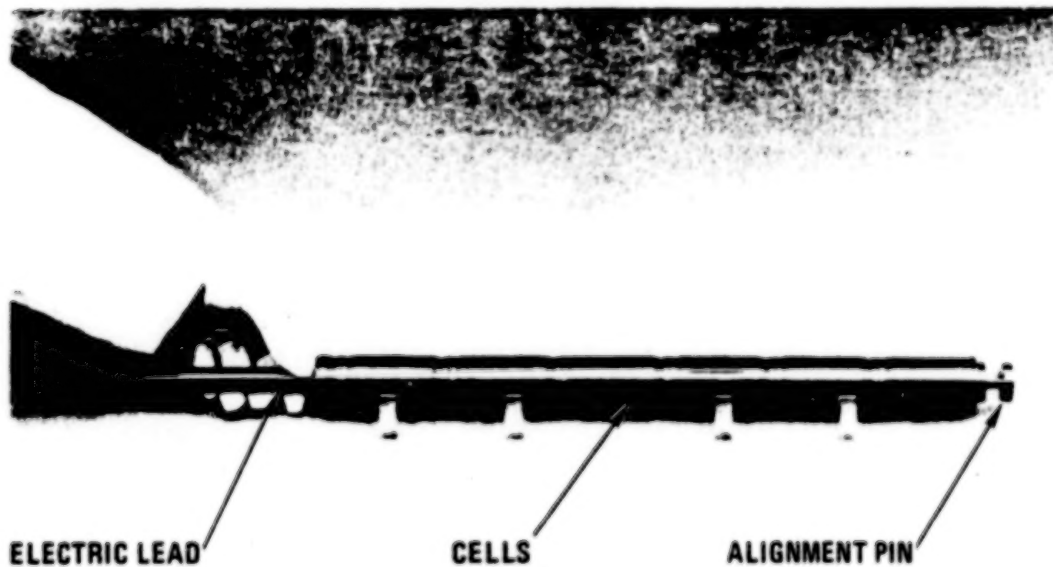


Figure 12 F.

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6F TFE SUBASSEMBLY PLASMA SPRAYED WITH Al_2O_3 ALONG WITH CESIUM RESERVOIR
AND SHEATH TUBE

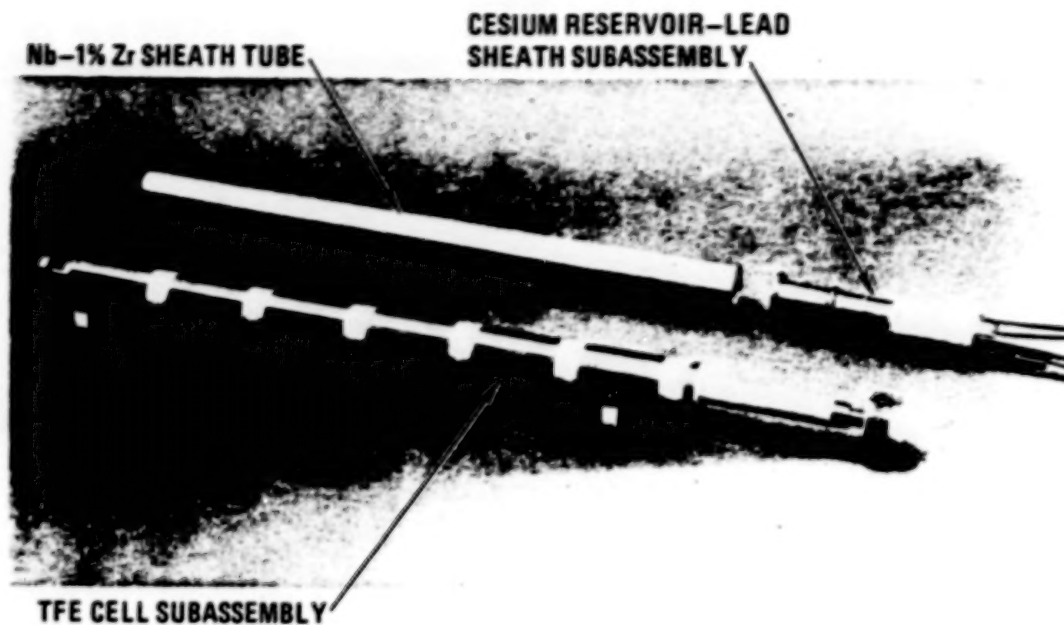


Figure 12G.

6F TFE READY FOR BONDING AT 1500 K FOR FOUR HOURS. TUNGSTEN WIRE IS SHOWN
WRAPPED OVER THE TRILAYERS

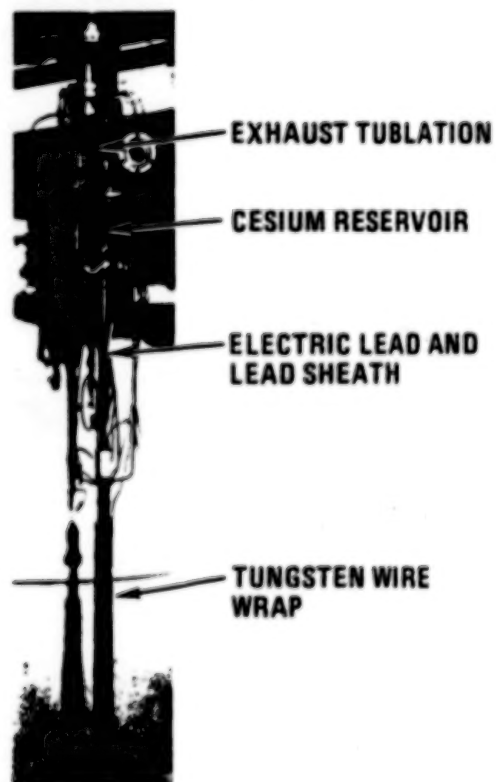


Figure 12H

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COMPLETED 6F TFE



Figure 12L.

**PERFORMANCE COMPARISON
OF NONFUELED AND UO_2 FUELED
THERMIONIC CELLS AND FUEL ELEMENTS**

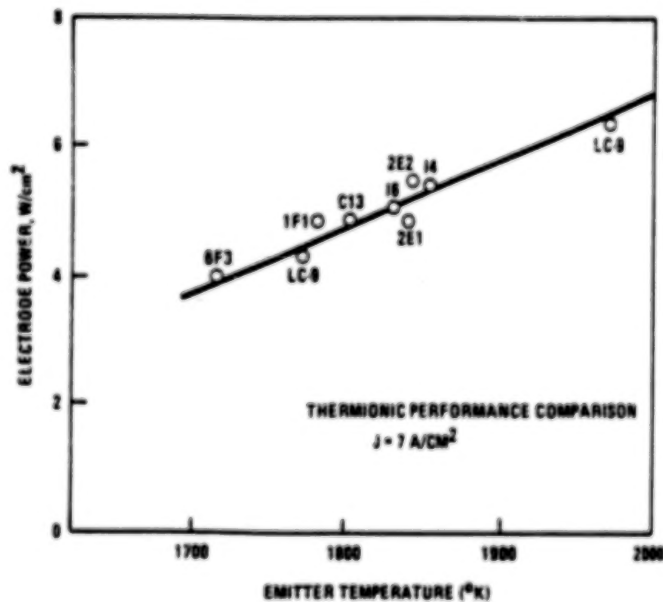


Figure 13.

MULTIPLE OPTIONS FOR TFE LIFETIME EXTENSION

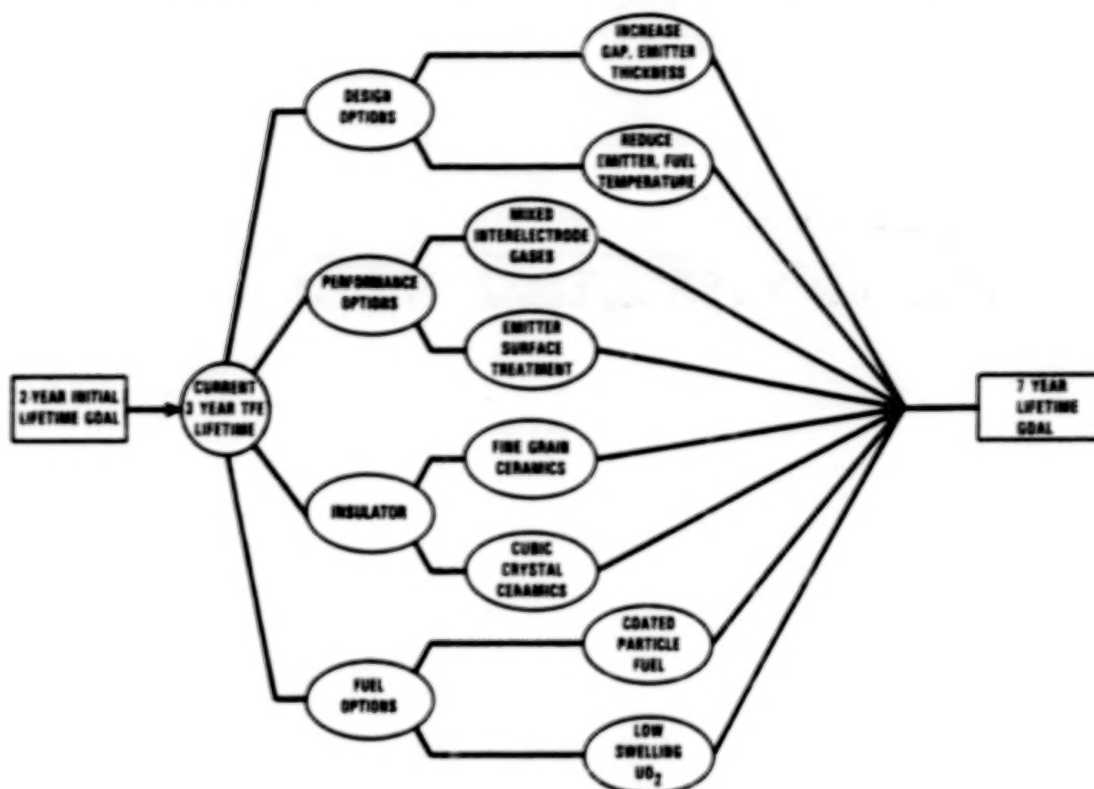


Figure 14.

HIGH EMITTER WORK FUNCTIONS REDUCE SENSITIVITY OF PERFORMANCE TO CHANGE IN INTERELECTRODE GAP

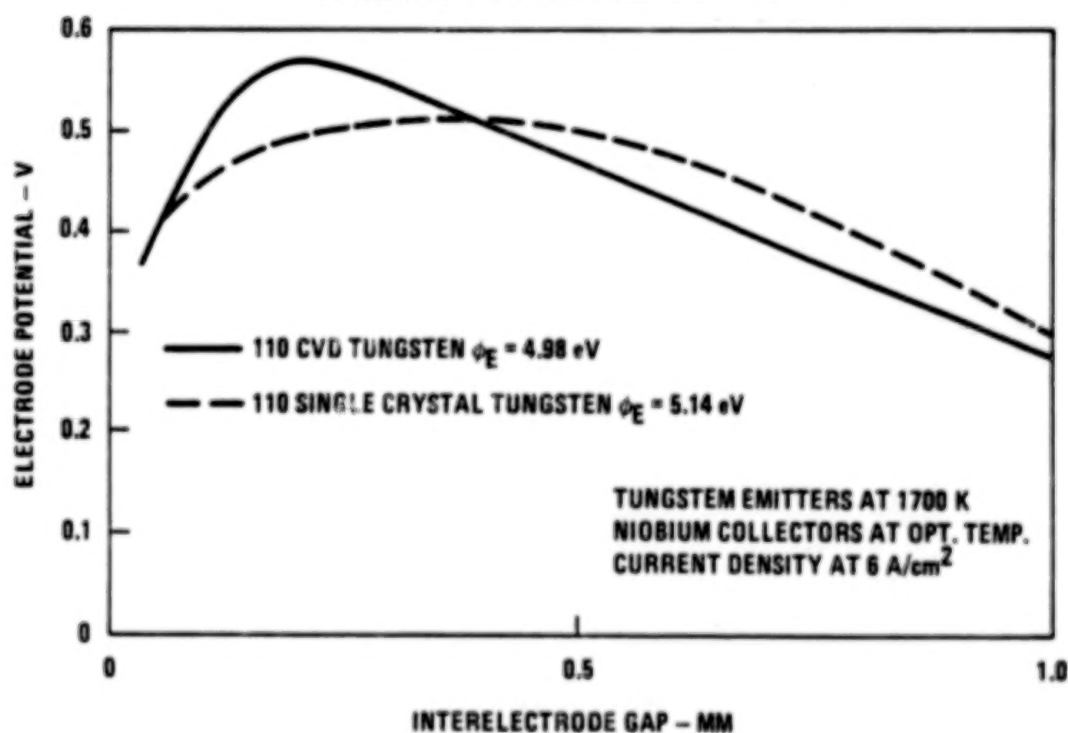


Figure 15.

EMITTER LIFETIME IMPROVEMENT OPTION NON-SWELLING COATED PARTICLE FUEL

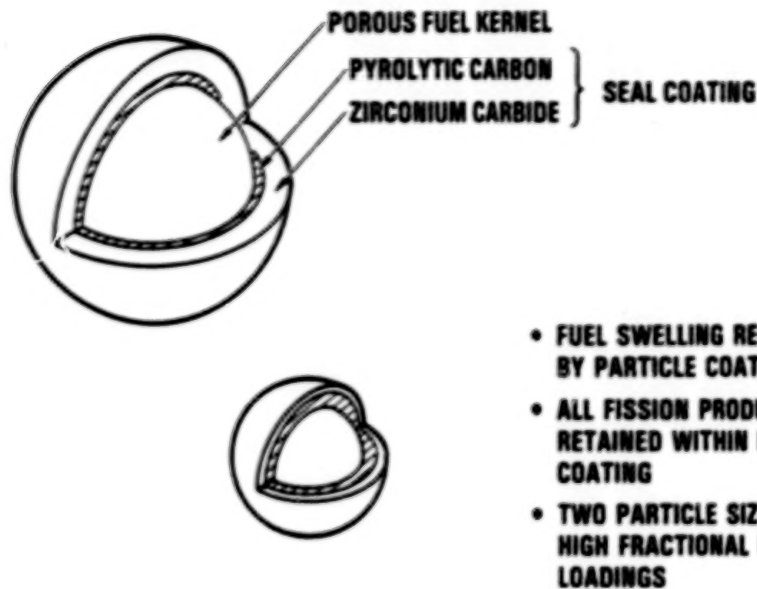
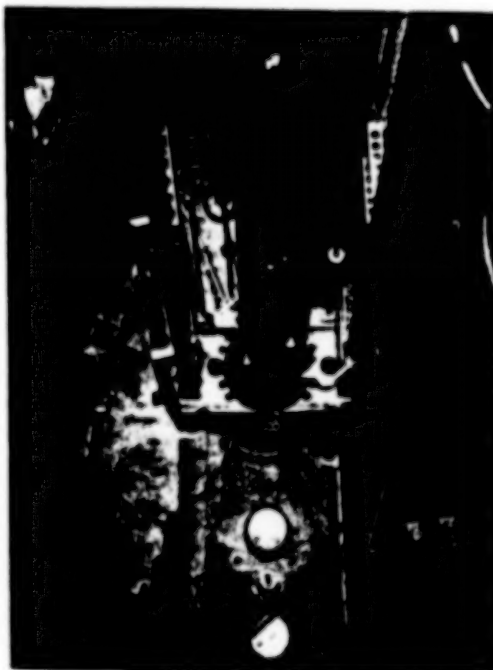


Figure 16.

TRIGA (TITR) TEST FACILITY IS AVAILABLE FOR CELL AND TFE TESTING



- ALLOWS TESTING OF 10-15 TFEs SIMULTANEOUSLY
- SIMULATES TFE REACTOR FAST NEUTRON FLUENCE
- CONTINUOUS INSTRUMENTED ON-LINE, REAL TIME TESTING
- ALLOWS NEUTRON RADIOGRAPHY OF TFEs
- HOT CELL FACILITIES ADJACENT TO TITR
- VERY ECONOMICAL \$1M/YR FOR FACILITY

Figure 17.

HIGH TEMPERATURE THERMOELECTRIC ENERGY CONVERSION

Charles Wood*
Jet Propulsion Laboratory
California Institute of Technology

The electronic and thermal transport properties of the most promising material systems for high temperature thermoelectric energy conversion are discussed from a theoretical and experimental viewpoint.

INTRODUCTION

Considerable advances were made in the late '50's and early '60's in the theory and development of materials for high-temperature thermoelectric energy conversion. This early work culminated in a variety of materials, spanning a range of temperatures, with the product of the figure of merit, Z , and temperature, T , i.e., the dimensionless figure of merit, ZT , of the order of one. This experimental limitation appeared to be universal and led a number of investigators to explore the possibility that a $ZT \sim 1$ also represents a theoretical limitation. It was found not to be so.

Research in the thermoelectric field has languished in the past two decades with only a small amount of research effort being conducted on high-temperature thermoelectric materials and most of that in Russia. The silicon-germanium alloys represent the current state-of-the-art in high temperature thermoelectric materials. These, like their lower temperature counterparts, the bismuth and lead chalcogenides, are exclusively broad-band semiconductors with conventional transport mechanisms, i.e., they conduct by itinerant motion of charge carriers. Examination of the equation for thermoelectric conversion efficiency (see below) shows that for a real breakthrough to occur, i.e., the achievement $ZT \gg 1$, much higher hot-junction temperatures are required in addition to high Z values. This naturally involves a search amongst the refractory materials for likely candidates and has led to exploratory research into new classes of high-temperature semiconducting materials. Recent work on these materials, i.e., the boron-rich borides and the rare-earth chalcogenides, will be reviewed after a brief introduction to the theory.

THEORY

The efficiency (η) for the conversion of heat to electrical energy using a thermoelectric couple with a hot junction at temperature T_1 and a cold junction at temperature T_0 is given by (ref. 1)

*The research described in this paper was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under contract with the National Aeronautics and Space Administration.

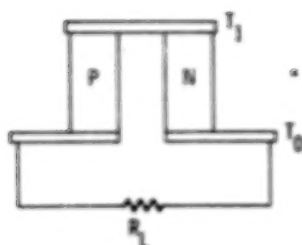


FIGURE 1. SEMICONDUCTOR THERMOCOUPLE

and Z is a materials parameter, called the figure of merit, which is related to the Seebeck coefficient (often misnamed the thermoelectric "power"), α_1, α_2 ; the electrical resistivity, ρ_1, ρ_2 , and the total thermal conductivity κ_1, κ_2 of the legs 1 and 2 of the thermocouple as follows:

$$Z = \frac{(\alpha_1 - \alpha_2)^2}{[(\rho_1 \kappa_1)^{1/2} + (\rho_2 \kappa_2)^{1/2}]^2} \quad (2)$$

The equation for conversion efficiency (η) shows that the thermodynamic efficiency of an ideal reversible engine (first term) is reduced by the irreversible losses of heat conduction ($\propto \kappa$) and Joule heating ($\propto \rho$) in the thermocouple (second term). Obviously, increasing the figure of merit (Z) and the hot junction temperature (T_1) increases the conversion efficiency; in the latter, by increasing both terms in Equation (1). For the purpose of comparing materials, it is convenient to define Z for a single material as

$$Z = \frac{\alpha^2}{\rho \kappa} = \frac{\alpha \sigma}{\kappa} \quad (3)$$

where σ is the electrical conductivity. The dimensions of Z are deg^{-1} and it is common to compare the properties of materials in terms of the dimensionless parameter ZT .

ITINERANT SEMICONDUCTORS

It is found that semiconductors have the highest Z values and are therefore the preferred class of materials for thermoelectric energy conversion. As illustrated in figure 2, metals have high electrical conductivity but the Seebeck coefficients are too low. Insulators can have high Seebeck coefficients but are too electrically resistive. A good compromise can be made by choosing highly-doped semiconductors with σ and σ values intermediate between the properties of insulators and metals.

Optimization

For a non-degenerate extrinsic semiconductor the Seebeck coefficient and electrical conductivity are related as follows:

$$\alpha = \pm \frac{k}{e} \left[A + \ln \left(\frac{N e \mu}{\sigma} \right) \right] \quad (4)$$

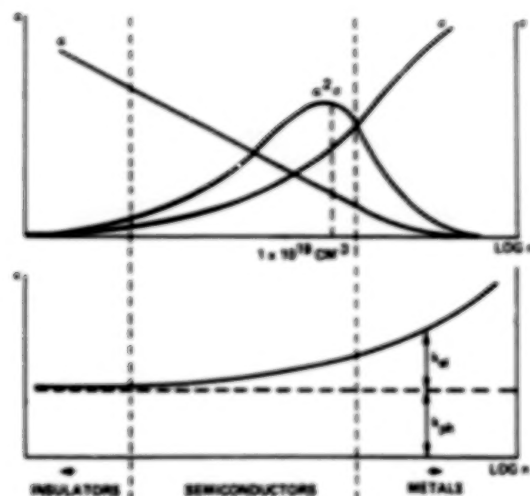


FIGURE 2. THE THERMOELECTRIC PROPERTIES OF METALS, SEMICONDUCTORS AND INSULATORS

where A is a constant whose value depends on the type of charge carrier scattering mechanism; μ is the mobility of the carrier of charge, e , and effective mass, m^* ; and N is the effective density of states:

$$N = 2(2\pi m^* kT/h^2)^{3/2} \quad (5)$$

Obviously as σ increases α decreases. However, it can be shown (ref. 1) that the numerator of Z , i.e., $\alpha^2 \sigma$, has an optimum value at a semi-degenerate concentration of charge carriers of about 10^{19} cm^{-3} . This value has to be modified slightly to allow for any significant contribution that the electronic conduction makes to thermal conductivity (κ_{el}), since in the denominator of Z , the total thermal conductivity

$$\kappa = \kappa_{ph} + \kappa_{el} = \kappa_{ph} + L\sigma T \quad (6)$$

where κ_{ph} is the lattice or phonon contribution and L is the Lorenz constant. Factoring in this term yields a maximum figure of merit of

$$Z_{\max} = \frac{A}{AT} \left(\frac{\kappa_{el}}{\kappa_{ph}} \right) \left(1 + \frac{\kappa_{el}}{\kappa_{ph}} \right) \quad (7)$$

At high temperatures a third term, κ_{rad} , should be included in Equation 6, i.e., radiative heat transfer may contribute to the total thermal conductivity. Dixon et al (ref. 2) have calculated the effect of heat transfer by radiation (photons) on lowering Z_{\max} . However, other investigators (refs. 3, 4) have shown that, at the high doping levels and effective mass values typical for thermoelectric materials, this effect is negligible up to $\sim 2000 \text{ K}$.

Since the optimum doping level corresponds to a semi-degenerate carrier concentration Chasmar and Stratton (ref. 5) and Rittner (ref. 6) used generalized Fermi-Dirac functions to calculate exact Z values for arbitrary degrees of degeneracy. They also included various types of charge-carrier scattering in their analysis. The magnitude of $(ZT)_{\max}$ and the depth of the Fermi-level below the conduction band at optimum doping both increase with the value of a material parameter:

$$\beta = 8.952 \times 10^{-6} (\mu/\kappa_{ph}) (T/300)^{5/2} m^{3/2} \quad (8)$$

A plot, derived from their curves, showing the variation of $(ZT)_{\max}$ with β for various values of scattering parameter, r , is shown in figure 3.

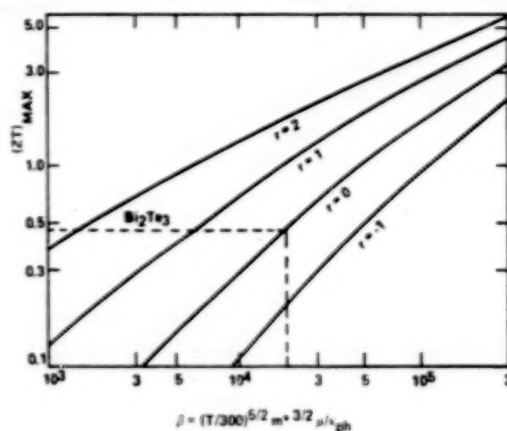


FIGURE 3. $(ZT)_{\max}$ VERSUS β FOR CHARGE CARRIER SCATTERING BY: IMPURITIES ($r = 2$), IONIC LATTICES ($r = 1$), ACOUSTIC-MODE LATTICE VIBRATIONS ($r = 0$), AND HIGHLY DISORDERED (AMORPHOUS) LATTICES ($r = 1$).

It is seen from (8) that Z can be related to fundamental material parameters (refs. 5-7), i.e.,

$$Z \propto m^{3/2} \mu / \kappa_{ph} \quad (9)$$

which suggests that the ideal thermoelectric material should satisfy the somewhat conflicting requirements of high effective mass, high mobility, and low lattice thermal conductivity.

By expressing α , σ and κ in terms of Fermi-integrals and substituting into Z it can be shown that the value of Z is approximately proportional to the scattering constant r (fig. 3). This suggests, as proposed by Rittner (ref. 6) and Joffe (ref. 1), that ionized-impurity scattering with $r = 2$ is the most desirable scattering mode.

Temperature Variation of Z

The exact temperature dependence of ZT can be determined from figure 3 by simply selecting the appropriate curve for the predominant scattering mechanism and substituting the temperature dependencies of μ and κ_{ph} in Chasmar and Stratton's dimensionless parameter β . For example, for acoustic-mode lattice scattering of charge carriers ($r = 0$), and assuming $\kappa_{ph} \propto T^{-1}$, gives $\mu \propto T^{-3/2}$ and, therefore, $\beta \propto T^2$. Whereas, ionized impurity scattering ($r = 2$) gives $\mu \propto T^{3/2}$ and $\beta \propto T^5$. However, these rapid increases of β and hence, ZT with increasing temperature will be severely limited at high temperatures when intrinsic conduction begins to play a role. The contribution to the conduction processes from charge of opposite sign will be subtractive in the Seebeck coefficient. Also, the thermal conductivity will rise

rapidly at the onset of significant charge-carrier recombination. The temperature at which either of these processes will start to make a significant contribution will depend primarily on the value of the band gap, the effective mass and the mobility ratios of the charge carriers.

An optimum band gap for a particular temperature of operation is apparent. In materials with large gaps the mobilities are often low with correspondingly low electrical conductivities. If the band gap is too small then, as indicated above, intrinsic conduction will set in. Criteria for optimum band gaps have been established by Chasmar and Stratton (ref. 5) assuming the transport properties similar to those of Bi_2Te_3 . Alternatively, ignoring the presence of minority carriers Aigrain (ref. 8) states that the temperature at which intrinsic conduction occurs, $T_i \sim E_g/2k$, where T_i can be equated to T_h , the hot-junction temperature. It should be noted that the energy gap, E_g , is the value at T_h , not room-temperature.

Thermal Conductivity

A simple method of estimating the thermal conductivity (κ) for any entity is to employ the classical kinetic theory of gases formula:

$$\kappa = 1/3 C_v l v \quad (10)$$

where C_v is the specific heat at constant volume, v is the velocity and l is the mean free path. Dugdale and MacDonald (ref. 9) expressed the phonon mean free path as $l \sim a/\xi Y T$ where a is the lattice constant, ξ is the expansion coefficient and Y is the Gruneisen constant and substituting in Equation (10) gives

$$\kappa_{ph} \sim \frac{1}{3} C_v \frac{a}{\xi Y T} v \quad (11)$$

Liebfried and Schlomann (ref. 10) derived the following relationship:

$$\kappa_{ph} \sim \frac{M a \theta_D^3}{T Y^2} \quad (12)$$

where \bar{M} is the average atomic mass, and θ_D is the Debye temperature. At first glance, it would be expected that a material of a high mean atomic mass would have high lattice thermal conductivity but the mass dependence of $(a\theta_D^3)$ predominates and κ_{ph} falls with increasing mass as shown in figure 4. The figure also shows a κ_{ph} dependence on mass difference and degree of ionicity in the bonding. Equations (11) and (12) can be shown to be equivalent (ref. 9).

EXPERIMENTAL RESULTS

As mentioned earlier the highest Z values are found in semiconducting materials. An inherent advantage of semiconductors is that they present a wide choice of elements and compounds for selection of properties suited to a particular application; e.g., such as thermoelectric energy conversion. In particular, they offer the flexibility of tailoring their electrical properties to this particular application.

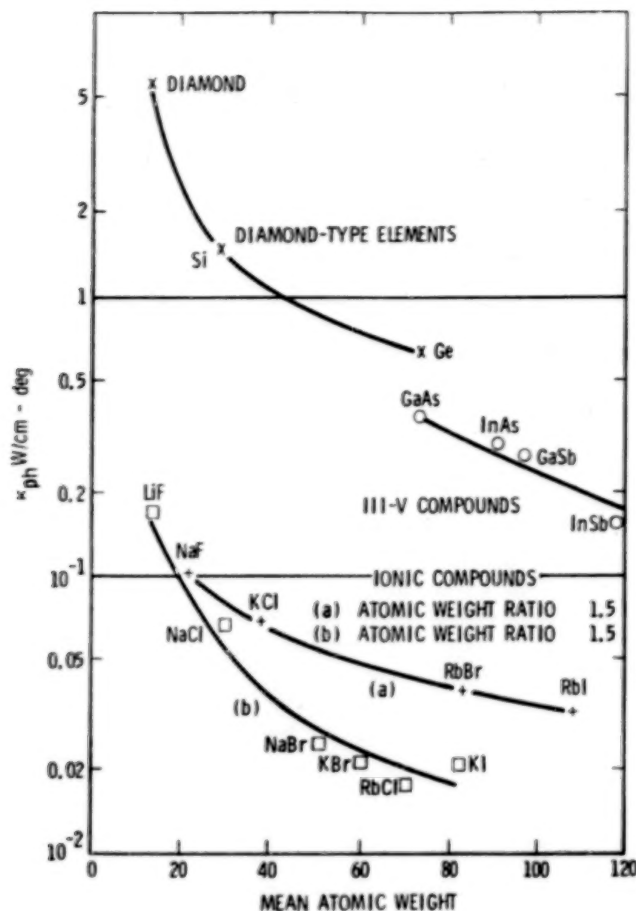


FIGURE 4. DEPENDENCE OF LATTICE THERMAL CONDUCTIVITY ON MEAN ATOMIC WEIGHT FOR COVALENT AND IONIC CRYSTALS

Guided by the theoretical work described above, a number of materials have been developed with respectable Z values for operation in the temperature range from room-temperature up to $\sim 1000^\circ\text{C}$. These are shown in figure 5 where it can be seen that each material has an optimum temperature range of operation. A plot of $ZT = 1$ is seen to envelope the peaks of these curves and led to the belief that $ZT \sim 1$ represents a limit for any material. Numerous attempts to determine a theoretical limit have been unsuccessful (refs. 11-15) and it seems safe to say that no theoretical limit on Z exists.

Examination of the equation for conversion efficiency (η) shows that considerable advantage can be obtained by operating the junction at higher temperatures. Hence the thrust of current research into thermoelectric energy conversion is toward high-temperature refractory materials. The question of stability of materials at high temperature involves not only high melting points but also low dissociation vapor pressures and, since it is essential that the electrical properties do not change, absence of phase-changes, precipitation of, and changes in, defect concentration.

One class of refractory itinerant semiconductors receiving considerable attention for high temperature thermoelectric energy conversion applications is the rare-earth chalcogenides.

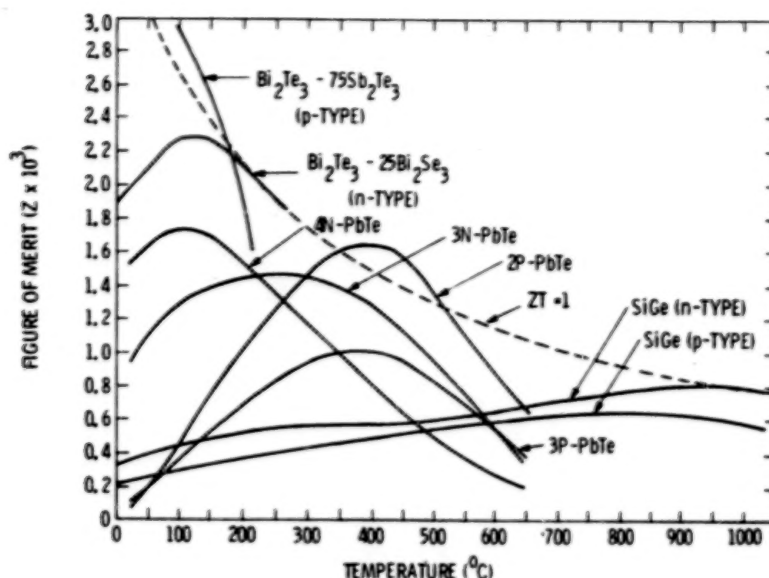


FIGURE 5. FIGURE OF MERIT (Z) OF SELECTED THERMOELECTRIC MATERIALS AS A FUNCTION OF TEMPERATURE

Rare Earth Chalcogenides

Rare-earth chalcogenides generally form the following binary compounds: RX , $R_3X_4 - R_2X_3$, RX_2 , where R represents the rare-earth and X the chalcogenide atoms S , Se , or Te . The RX compounds crystallize in a fcc NaCl structure, the $R_3X_4 - R_2X_3$ form solid solutions in a bcc (Th_3P_4) tetragonal or orthorhombic structure, and the RX_2 compounds usually are found in either cubic or tetragonal structure (ref. 16).

Since RX_2 readily dissociates by evolution of the chalcogen at temperatures well below the melting point, and RX compounds are generally too metallic, only compositions in the homogenous range $R_3X_4 - R_2X_3$ have been extensively investigated for high-temperature thermoelectric applications. There are often two and sometimes three polymorphic modifications of the solid solutions $R_3X_4 - R_2X_3$: (i) a low-temperature ($< 900 - 1000^\circ C$) orthorhombic α -phase; (ii) an intermediate temperature (900 to $1300^\circ C$) tetragonal β -phase; and (iii) high temperature ($> 1300^\circ C$) cubic (Th_3P_4) metastable γ -phase (ref. 16). It is not clear from the literature whether these polymorphic forms occur over the whole compositional range from R_2X_3 to R_3X_4 . Furthermore, with the exception of La_2S_3 it is not yet resolved as to whether β -phase R_2X_3 is truly a polymorph or is representative of a ternary compound involving oxygen (ref. 17).

In the cubic Th_3P_4 structure there are 28 lattice sites comprising 12 rare-earth atom sites each having eight chalcogen atom neighbors and 16 chalcogen atoms each having six rare-earth neighbors. It is convenient to designate the unit cell as $4(R_{3-x}V_xX_4)$ where V is a rare-earth vacancy. For R_2X_3 compounds $1/3$ of the 12 rare-earth sites are vacant, i.e., $x = 1/3$, and for the R_3X_4 compounds there are no vacant sites, i.e., $x = 0$. The ionic character of the lattice can be described by the formula $(R^{3+})_{3-x}V_x(X^{2-})_4(e^{-1})_{1-3x}$. Starting with R_2X_3 , the two R^{3+} ions contribute six electrons to the chemical bonding which are taken up by the three X^{2-} ions. Thus, there are no excess electrons available for conduction and all R_2X_3 compounds are insulators. As additional rare-earth atoms are added, Re^{3+} ions are introduced at random vacancy sites in the lattice thus contributing three electrons

per ion.

At the other extreme, in R_3X_4 the three R^{3+} ions contribute nine bonding electrons while four X^{-2} ions accept only eight electrons giving rise to an excess of one conduction electron per formula unit. The concentration of $R_{3-x}X_4$ units is $N_A d/M$ ($=6.25 \times 10^{21} \text{ cm}^{-3}$ for Ce_3S_4) where N_A is Avogadro's number, d is the density and M is the molecular weight. Thus, the perfect crystal with no vacancies ($x = 0$) has a free electron concentration of this value ($6.25 \times 10^{21} \text{ cm}^{-3}$ for Ce_3S_4). The presence of x vacancies requires $(1 - 3x)$ electrons in the conduction band. Hence, in going from R_2X_3 to R_3X_4 , there is a transition to metallic conduction as the vacancies are filled with rare-earth ions.

Since stoichiometric R_2X_3 compounds are insulators, most materials examined for thermoelectric applications are hot-pressed powders or single crystals of $R_{3-x}X_4$ with $x < 1/3$, i.e., varying degrees of excess of rare-earth atoms. These stoichiometries have been found to yield degenerate wide band-gap semiconductors, with electrical conductivities of $\sim 10^3 \text{ ohm}^{-1} \text{ cm}^{-1}$, carrier concentrations greater than $\sim 10^{20} \text{ cm}^{-3}$ and charge carrier mobilities of ~ 1 to $10 \text{ cm}^2/\text{volt sec}$. Unfortunately, the polymorphic form being studied and reported upon, is not always specified, the samples are generally not known to be single-phase and the degree of oxygen contamination is often unknown, all of which are central to the thorough understanding and utilization of these compounds.

Thermal Conductivity

Lattice thermal conductivities (κ_{ph}) of the $R_{3-x}X_4$ compounds generally have low values in the range 0.005 to 0.01 watt/cm-deg. (ref. 18) which are primarily a result of three factors. First, the Debye temperatures (θ_D) are low (~ 200 to 400 K) and κ_{ph} is proportional to θ_D^3 . Secondly, these are complicated crystal structures with a fairly large number of atoms (28) per unit cell. Since there are three acoustic modes per unit cell the number of optic modes, $N = 28 - 3$. Examination of the dispersion curve, ω versus k , shows that these optic modes of vibration have very low propagation velocities. Therefore, the majority of phonons are optical and have low velocities. Thirdly, the rare-earth ion vacancies produce mass fluctuations which, in addition to the mass difference between the rare earth and chalcogen ions, enhances phonon scattering.

Transport Mechanisms

From the above discussion it appears that rare-earth chalcogenides behave as highly-degenerate n-type semiconductors over most of the composition range R_3X_4 to R_2X_3 . For compositions very close to R_2X_3 , hence, low carrier concentrations, or at extremely high temperatures, the degree of degeneracy diminishes to a semi- or non-degenerate condition. At R_2X_3 the compound is an insulator. The transport properties can be described by the equations for almost-free electrons, or degenerate semiconductors, provided x does not approach $1/3$. However, the conduction band does appear to be narrow from the large values of effective mass and the small values of the mobility. Cutler et al. (ref. 19) have suggested that from the similarity of behavior of praseodymium sulfide and cerium sulfide that it is the outer 5d and 6s electrons that are involved in the conduction process and not the inner shell 4f electrons. However, group-theoretical calculations by Goryachev et al. (ref. 20) show that the conduction band is more complex, comprising a mixture of 6s, 5d and 4f states. There also appears to be localized levels of d and f character with high

density of states. If these states produce a large gradient in the density of states $g(E)$ at the Fermi-surface, then the Seebeck coefficient, which is proportional to $d[\ln(\mu g)]dE$ (ref. 21), can be large.

Since the rare-earth chalcogenides are so similar in properties it is of interest to consider which specific rare-earth compound has the highest potential thermoelectric performance. Ignoring for the moment thermal conductivity differences (they appear to differ very little from each other between the various rare-earth chalcogenides) examination of Equation (9) points the way to selecting the best compositions. The room-temperature physical properties of a large number of rare-earth chalcogenides have been tabulated by Zhuze et al. (ref. 22). Their data has been employed to calculate values for $m^{*3/2}\mu$ listed in table I.

TABLE I. Properties of Rare Earth Chalcogenides (R_3X_4) at Room-Temperature

Compound	Mobility, $\mu(\text{cm}^2/\text{v-s})$	Effective Mass, m^*	$m^{*3/2}\mu$
La_3S_4	3.5	3.6	23.9
Ce_3S_4	3.1	2.8	14.5
Nd_3S_4	3.2	2.7	14.2
Pr_3Se_4	2.6	2.6	10.9
La_3Te_4	11.5	1.8	27.8
Ce_3Te_4	4.2	2.1	12.8
Pr_3Te_4	6.2	1.6	12.5
Nd_3Te_4	5.0	2.0	14.1

It is seen that $\text{La}_{3-x}\text{Te}_4$ has the greatest potential for high Z values. This is supported by the work of Golikova and Rudnik (ref. 23) who claim that La_3Te_4 has the a value of $0.91 \times 10^{-3} \text{deg}^{-1}$ and, hence, a ZT of 1.41 at 1550 K. Zhuze et al. (ref. 22), reporting essentially the same data, quote a Z of 0.53×10^{-3} at 1400 K for the nominal composition La_3Te_4 . This and thermoelectric data reported on other rare-earth chalcogenides is illustrated by the plot of ZT versus T in figure 6.

Although the transport properties for compositions close to and including R_3X_4 have been elucidated further work appears necessary in order to clarify the transport mechanism near the composition R_2X_3 . Golikova and Rudnik (ref. 23) state that as $x \rightarrow 1/3$, i.e., R_2X_3 compounds, hopping-type conductivity prevails, although no details are given. Earlier, Cutler and Leavy (ref. 28) made a detailed study of the electronic transport properties of high resistivity $\text{Ce}_{3-x}\text{S}_4$ samples. Hall effect measurements as a function of temperature showed that for samples with $x = 1/3$ the mobility was indeed activated. They interpreted their data as indicating two extreme conduction processes to be operative (i) a hopping process with electrons in localized states (these states were proposed to arise from fluctuations in potential resulting from random lattice vacancies) and (ii) motion of non-localized electrons. The $\text{Ce}_{3-x}\text{S}_4$ compounds exhibit a continuous variation between the two extremes depending upon composition and temperature. The data of Taher et al. (ref. 29) on Y-phase R_2X_3 compounds presumably could be fitted to this model. Small polaron hopping thus presents a model to explain the data on high resistivity R_2X_3 compounds.

Summarizing, it appears that it is possible to realize values for ZT of greater than one at high temperatures in the rare-earth chalcogenide system. In nearly every case, however, to arrive at these high ZT values the thermal conductivity was either estimated from values for other rare-earth chalcogenides and/or the data was

extrapolated from low temperatures. Accurate thermal conductivity data are difficult to obtain, particularly at high temperatures and so the quoted values of ZT should be viewed with some reservation. Notwithstanding, considerable room for improvement of performance exists at lower temperatures, -800 to 1300 K, i.e., in the temperature range where the Si-Ge alloys are pre-eminent.

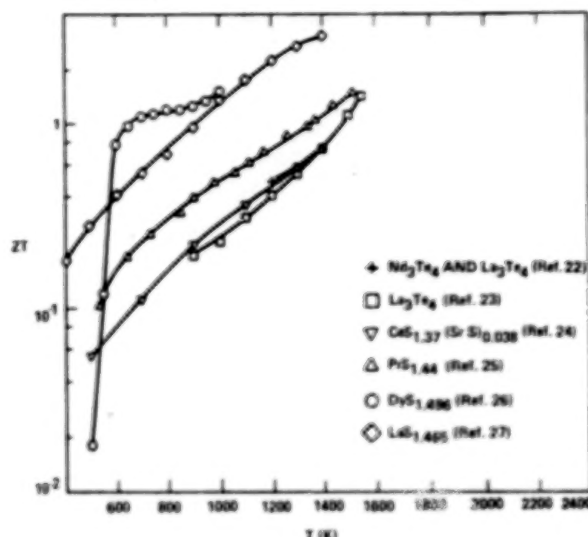


FIGURE 6. ZT VALUES FOR RARE-EARTH CHALCOGENIDES

In addition, although the dissociation vapor pressures of the rare-earth chalcogenides at high temperatures are surprisingly low for sulfur, selenium and tellurium compounds, the phase stability is still in question. Ryan et al (ref. 24) ascribed changes in thermoelectric properties on high-temperature cycling to oxygen contamination. This was discounted by Cutler et al (ref. 25) whose studies pointed to disproportionation of high-temperature γ -phase of $Ce_{3-x}S_4$ into a sulfur-rich intermediate temperature β -phase in equilibrium with a sulfur-poor γ -phase. Considerable improvement in phase stability was obtained by doping with the alkaline earths, Ba, Ca and Sr. It was conjectured that this improvement resulted from a filling of vacancy sites. Obviously, more phase, mass and lattice defect stability studies are needed before these materials can be safely employed in thermoelectric applications for extended periods at high temperatures.

HOPPING-TYPE SEMICONDUCTORS

In the discussion so far we have considered materials with conventional semi-conducting transport properties, i.e., ones which conduct by the itinerant motion of charge carriers. However, there is another broad class of materials which conduct by the charge carriers hopping from site to site.

Ure and Heikes (ref. 30) have derived expression for the transport properties of narrow band or mixed valency semiconductors in which charge carriers are trapped by polaron formation and transport occurs by charge hopping from site to site. The following expression for the electrical conductivity was derived:

$$\sigma = \frac{c(1-c)e^2}{akT} \tau_0^{-1} \exp \left(-\frac{\Delta G}{kT} \right) \quad (13)$$

where a is the diffusion length or distance between sites, assumed to equal the lattice constant; c is the charge carrier concentration; τ_0^{-1} the a priori transition probability at infinite temperature which is related to the lattice vibrational frequency; and ΔG is the hopping activation energy. The Seebeck coefficient was given by

$$\alpha = \frac{k}{e} \left[-\epsilon \frac{\Delta H_R}{kT} - \ln \left(\frac{c}{1-c} \right) \right] \exp \left(\frac{-\Delta G}{kT} \right) \quad (14)$$

where ΔH_R is the change in energy associated with the removal of a charged particle and ϵ is related to the lattice force constants at a site. Experimentally ϵ is of the order of one. Hence,

$$ZT = \frac{k}{\kappa_{ph} a} \left[\frac{\Delta G_R}{kT} - \epsilon \frac{\Delta H_R}{kT} - \ln \left(\frac{c}{1-c} \right) \right]^2 \tau_0^{-1} c (1-c) \exp \frac{-\Delta G}{kT} \quad (15)$$

Optimization

In order to maximize (ZT) the coefficient and the polaron binding energy G_R were set equal to zero by Ure and Heikes (ref. 30), thus

$$ZT = \frac{k}{\kappa_{ph} a} \left[\ln \left(\frac{c}{1-c} \right) \right]^2 c (1-c) \tau_0^{-1} \quad (16)$$

Differentiation of this equation with respect to c yields a maximum at $c \sim 0.1$. However, this value should be accepted with reservation. Ure and Heikes state that the assumption $\epsilon = 0$ is not born out experimentally. Furthermore, setting $\Delta G = 0$ raises the electrical conductivity but lowers the Seebeck coefficient. Thus, setting the term $\epsilon \Delta H_R / kT = 1$ in the Seebeck coefficient and optimizing ZT leads to a c value closer to 0.2.

Temperature Variation of Z

In materials which exhibit hopping-type conductivity, the charge carrier mobility and, hence, the electrical conductivity increases with the increase in temperature. If the conduction mechanism is by small polaron hopping between sites which are equivalent in energy then the Seebeck coefficient should be roughly independent of temperature.

In a disordered small polaron material with charge carriers hopping between inequivalent sites then vibrational energy may be transported with the carrier as it hops. This will give rise to a temperature dependent term in the Seebeck coefficient (ref. 31) i.e.,

$$\alpha = (T\Delta S + E_T)/qT \quad (17)$$

where the first (standard) term is proportional to the average change of entropy, ΔS , of the material when a charge carrier is injected into it and the second term is proportional to the average vibrational energy transported with a carrier as it hops, E_T . This additional contribution to the carrier produced heat flow enhances the Seebeck coefficient. Thus, in a small polaron material, in addition to the electrical conductivity rising with temperature the Seebeck coefficient may also increase with temperature. Consequently, Z will increase drastically with temperature and can be considerably greater than the limit predicted by Ure and Heikes

(ref. 30) at high temperatures. This increase in Z may, however, be tempered somewhat by the carrier produced heat flow now introducing a κ_{el} contribution which is not present in the absence of vibrational energy transport.

EXPERIMENTAL RESULTS

Examples of materials which fall into the class of hopping-type semiconductors are the boron-rich borides, the alkali-halides and the transition metal oxides. We will consider below the performance of some boron-rich borides as high temperature thermoelectric materials.

Boron and Borides

Boron forms refractory compounds with a large number of elements and several of these compounds have been investigated for high-temperature thermoelectric conversion. Elemental boron is highly refractory (m.p. $\sim 2500^\circ\text{C}$) and its thermoelectric properties have also been studied (ref. 32). Boron is generally accepted to have at least three and possibly four allotropic forms; α -rhombohedral ($R\bar{3}m$, $a = 5.075 \text{ \AA}$, $\alpha = 58^\circ 06'$); β -rhombohedral ($R\bar{3}m$, $a = 10.145 \text{ \AA}$, $\alpha = 65^\circ 17'$); α - or I - tetragonal ($P4_2m$, $a = 8.75 \text{ \AA}$, $c = 5.06 \text{ \AA}$) which may not correspond to an allotropic form of pure boron [28] and β - or II- or III- tetragonal ($P4_122$, $a = 10.12 \text{ \AA}$, $c = 14.14 \text{ \AA}$). Boron can also occur in the amorphous form. Many boron-rich borides are structural analogs of one of the four boron crystalline modifications. These structures are characterized by an arrangement of an icosahedral cluster of B_{12} atoms bound either to one another or to isolated atoms by directed bonds. Excellent surveys of the properties of boron and borides have been published by Golikova (ref. 32), Matkovich (ref. 33), and Samsonov et al. (ref. 34).

Only four materials have been investigated in any depth for thermoelectric applications, i.e., β -boron (105 atoms/unit cell), $B_{14}Si$ (isostructural with β -boron and with 105 atoms/unit cell), B_xC (an analog of α -boron with 15 atoms/unit cell) and α - AlB_{12} (same space group as β - or II-tetragonal boron with from 187 to 208 atoms/unit cell). Thermoelectric data on these materials are shown in figure 7.

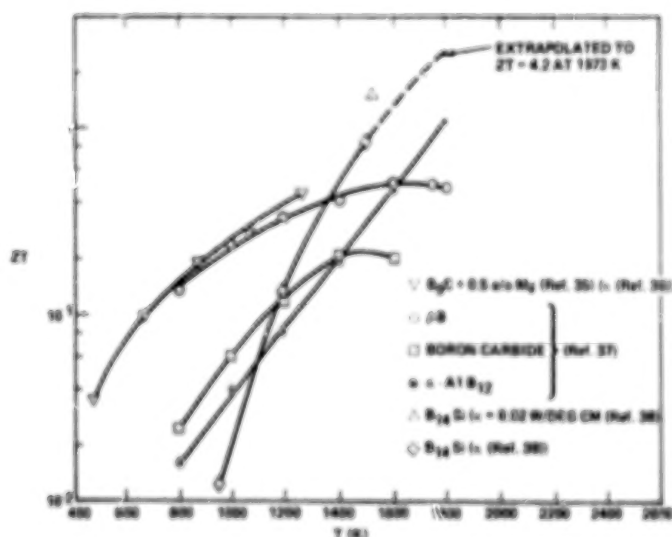


FIGURE 7. ZT VALUES FOR BORON-RICH BORIDES

Thermal Conductivity

The surprisingly low values for the high temperature thermal conductivity of boron and boron-rich borides has been attributed to the complexity of the crystal structure (refs. 40, 41). At low temperatures the dominant contribution to the thermal conductivity is due to long-wavelength acoustic phonons. The contributions to the specific heat by the acoustic modes are associated with the vibrations of the unit cell as a whole. Consequently, this contribution should vary inversely proportional to the number of atoms/unit cell (N_c). Thus, the magnitude of should vary inversely with N_c supposing the velocity and mean free path to change insignificantly from one boride to another. In addition, the acoustic branches are cut-off at smaller values of wavevector while the number of optical vibrations increase as N_c increases. In β -boron for example with 105 atoms/unit cell there are 3 acoustic branches and 312 optical branches in the dispersion curve. As the temperature is increased the relative contribution of the high frequency (optical) modes to the thermal conductivity increases and can exceed the contribution by the acoustic modes. The group velocity of optical vibration is much lower than the sound velocity and so that their contribution to the thermal conductivity is low. Because of the structural complexity of boron and its analogues, groups of atoms in equivalent positions are widely separated. For an optical wave to propagate a definite phase relation must exist between these like groups of atoms. Perturbations arising from other groups of atoms in the path of propagation cause a breakdown of phase relationships i.e., independent local vibrations occur. Thus, the nature of phonon propagation in boron-rich borides closely approaches that in amorphous solids with correspondingly low values. Not all boron-type structures appear to conform to this hypothesis--the thermal conductivity of α -boron (ref. 37) being a case in point that warrants further verification.

Transport Mechanisms

Boron and all of the boron-rich borides described above display a notable characteristic behavior, i.e., they exhibit a rising α and σ with increasing temperature over a wide temperature range. This has been generally interpreted as due to the hopping of charge carriers; a conclusion often supported by measurements showing that the mobility increases as the temperature increases. However, the models proposed to account for this conduction mechanism have differed from material to material or in some instances for the same material but over different temperature ranges.

Despite some differences between interpretations of the experimental data most investigators (refs. 32, 42, and 43) appear to have arrived at the conclusion that hopping occurs between high densities of localized states which arise naturally in the structure of β -boron. Berezin et al. (ref. 43) and Golikova et al. (ref. 44) have invoked an amorphous concept to explain the origin of these localized states. Because of the complicated structure of β -boron with a large unit cell of 105 atoms there are groups of non-equivalent atoms in the lattice having different coordination numbers, e.g., 13% of the atoms have coordination numbers of 8 and 9. These like-atoms are spaced about 10 interatomic distances apart and, thus, to a first approximation, there is no long range order. It is these atoms which are supposed to give rise to the localized levels.

The above discussion is also applicable to the boron-rich borides. Close parallels have been drawn between the properties of β -boron and α - AlB_{12} (ref. 44) and to a lesser extent B_{1-x}C_x (ref. 45) although some strong evidence has recently

been accumulated in support of small polaron-hopping in $B_{1-x}C_x$ compounds (ref. 31).

An alternative hypothesis has been proposed for the origin of the hopping process in $B_{14}Si$ which invokes a medium-range disorder model (ref. 46). The experiments conducted so far on $B_{14}Si$ have been performed solely on chemically vapor deposited specimens. Cast specimens of B_9Si exhibited an entirely different behavior with much higher mobilities (ref. 47). The question naturally arises as to how much the results on $B_{14}Si$ are influenced by the nature of the deposit or whether the cast specimens of B_9Si were single-phase.

It is of considerable interest and value to speculate on the origin of the hopping-type conduction in boron-rich borides since a thorough understanding of the mechanism will allow the tailoring and optimization of these materials for high-temperature thermoelectric and other applications. However, it may be more a matter of semantics in attempting to distinguish between the various models: localized states, small polaron hopping, medium range disorder, etc., when the charge carriers are strongly localized.

In summary, as with the rare-earth chalcogenides, ZT values in excess of one appear realizable at high temperatures. More work is obviously needed to extend the high Z values down to lower temperatures. The achievement of this objective will be greatly aided by a better understanding of both the electronic and thermal transport properties.

CONCLUSIONS

The excellent theoretical work carried out two decades ago adequately explained the transport behavior and effectively guided the development of thermoelectric materials of high conversion efficiencies. The more significant contributions involved the estimation of optimum doping concentrations, the reduction of thermal conductivity by solid solution doping and the development of a variety of materials with $ZT \sim 1$ in the temperature range 300 K to 1200 K. It was also shown that $ZT \sim 1$ is not a theoretical limitation although, experimentally, values in excess of one were not achieved.

In the subsequent years work has continued with emphasis on high temperature energy conversion and a number of promising materials have been discovered with indications that $ZT > 1$ is realizable. These materials can be divided into two classes: (i) the rare-earth chalcogenides with high vacancy concentrations, which behave as conventional highly-degenerate n-type semiconductors at room-temperature, with the degree of degeneracy decreasing to partial degeneracy at higher operating temperatures; and (ii) the boron-rich borides, which exhibit hopping p-type conductivity behavior.

It is not clear why the rare-earth chalcogenides are generally n-type and the boron-rich borides and β -boron generally p-type semiconductors, but it could be related to large mobility ratios of majority and minority charge carriers. It would be desirable, from the view of materials compatibility, if the thermocouple could be fabricated from n- and p-legs of the same compound. However, it appears to be extremely difficult to dope rare-earth chalcogenides p-type or boron-rich borides n-type.

Obviously, a considerable amount of work still needs to be done in order to

relate both the thermal and electronic transport to the composition and crystallographic structure in these classes of materials—particularly the identification of the hopping centers in the case of borides. Current theory is inadequate to describe the conditions for optimization of the thermoelectric Figure of Merit in materials which exhibit hopping conductivity and further theoretical work is needed. Since, in this class of materials, the conventional relationship between Seebeck coefficient and electrical conductivity is violated in a manner favorable to the achievement of high Z 's, the importance of further basic research into this area should not be overlooked. Hopefully this report will stir some further interest in this area or more generally into the transport properties of refractory materials.

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THERMAL MANAGEMENT TECHNOLOGY STATUS

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Beginning with Gemini and extending through the Shuttle program, active thermal management of manned spacecraft has been accomplished utilizing pumped single-phase fluid systems for acquisition, transport, and rejection of waste heat. These systems have proved to have significant reliability for the limited duration missions for which they were designed. However, the mission scope has expanded with the recently announced United States goal to establish the permanent presence of man in space by placing a Space Station in low earth orbit within a decade.

Preliminary Space Station planning has indicated large scale growth from the state-of-the-art Shuttle thermal management system, capable of rejecting approximately 30 kW, up to a fully operational Space Station capability of almost 300 kW. This growth in total thermal load is accompanied by an increase in electronic equipment thermal density, longer thermal transport distances, a requirement for more efficient use of waste heat, accommodation of various payloads, sequential system growth capability, and requisite long life/high reliability. In order to meet these requirements, a more sophisticated technology becomes necessary.

Through the NASA Manned Space Station Steering Committee Thermal Working Group, comprised of representatives from each of the NASA centers, a thermal technology plan has been generated. This plan presents a direction for thermal technology development for manned space vehicles and aids in the allocation of tasks and resources for the accomplishment of the planned evolution. Directions which have been charted for immediate activity have been divided into three distinct technology goal groups: (1) long life heat rejection, (2) versatile thermal acquisition and transport, and (3) integrated thermal utility. Representative elements of this comprehensive plan include the development of high capacity heat pipe radiators, a constructable radiator system, a centralized two-phase thermal bus, high capacity evaporator/condenser cold plates, and an integrated thermal analysis capability.

Space power system components will be utilizing and, to some extent, driving these developing technologies. Waste heat from power generation/distribution equipment must be dissipated or incorporated into the thermal management system. This equipment may be compatible

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with existing thermal control schemes or may require dedicated thermal systems. For example, an emerging power system requirement for high temperature ($>100^{\circ}\text{C}$) energy transport and rejection appears to be growing in visibility. This would initiate an additional thermal technology direction and development activity.

There are, however, certain critical technological barriers which will have to be surmounted in order for advanced thermal technologies (i.e., two-phase technology) to be of benefit to large space platforms or high power energy conversion systems. Integrated two-phase thermal management has established itself in proof-of-concept studies, but must be tested in a microgravity environment on a larger scale to generate confidence in all aspects of its utilization.

Transient phenomena in high temperature (liquid metal) heat pipes and thermal transport loops during cold start or power turn-down is not well understood, and must be for incorporation into a space vehicle thermal management system. Also, high temperature waste heat utilization must be defined in order to better delineate its integration into the thermal control network.

Thermal management technology for manned space systems is currently undergoing an evolution, but this evolution must be controlled and directed in a manner such that maximum benefits are derived from expended efforts. A focused view of future requirements and goals and a well defined path to their attainment is being mapped by the NASA organization.

INTRODUCTION

Inherent in dynamic energy conversion system operating cycle is the utilization of the latent heat of vaporization, as well as the sensible heat of a working fluid. The efficiency of the conversion process is greatly enhanced by using the expanded range of a fluid's thermal capacitance as it undergoes a phase change. Though this is an integral component of power cycle design for converting heat into work, this phenomenon has not yet been well utilized, particularly in space systems, for its thermal energy transport capability. Problems with pumping a two-phase fluid and in analytically predicting flow regimes and heat transfer coefficients have been the inhibitors in this technology incorporation. Empirically determined mathematical relationships for two-phase fluid flows have been accomplished for earth-bound utilization, but these are not yet available for microgravity multi-phase flow characterizations.

Therefore, the current state-of-the-art in manned spacecraft active thermal control systems (ATCS) is the use of single-phase fluids in closed-loop systems and liquid vaporization in open-loop systems. These take form in the Shuttle Orbiter vehicle as a mechanically-pumped closed water loop in the crew cabin which rejects its heat load to a mechanically-pumped freon radiator loop. If the heat load is too large for the radiators to handle, or if the radiators are not deployed, a supplemental open-loop flash evaporator system (FES) is used to cool the freon by vaporizing water into space. Below approximately 30 km,

during liftoff and entry portions of a Shuttle mission, an ammonia boiler system (ABS) instead of the FES, is used to reject the freon heat load. This system has performed adequately aboard the Orbiter vehicle and has demonstrated the capability to reject approximately 30 kw of thermal energy under favorable cold-sink conditions.

However, the thermal control requirements of a large Space Station stretch the capabilities of a single-phase fluid system and the mission scenario of long orbit duration opens the door for expanded heat transport technology. This technology expands on the mechanisms of heat pipe operation which were developed in the early 1960's and moves into large scale capillary-pumped loops, thermal utility buses, and space-constructable heat pipe radiators.

MOTIVATIONS FOR ADVANCED THERMAL TECHNOLOGY

The NASA-JSC experience with the Orbiter single-phase active thermal control system has illustrated the competency of the design but has also delineated its limitations. The majority of waste heat from electronics, fuel cells, human metabolism, etc. is dissipated to space through the heat rejection system (radiators, FES, or ABS). Though circumventing the problems caused by excessive subsystem integration and interaction, it is an inefficient system that results in little utilization of waste heat and prodigious use of electric heaters.

Inherent in the single-phase water coolant loop, as designed for the Orbiter, is a large temperature rise (from approximately 10°C to 32°C) through the equipment coldplates. This implies proper equipment ordering in the loop to maintain required temperature control ranges and the cognizance of upstream equipment intermittent operation temperature affects. Furthermore, the growth capability of the thermal control system is constrained due to the impact that additional equipment would have on downstream temperature ranges. Payloads are likewise limited in their placement in the loop and would be required to make/break fluid connections to tap into the system.

With the large thermal transport loads and distances which would be a part of a Space Station design, pumping requirements become quite large, driving up weight, cost, pump power consumption, and decreasing long-term reliability. Increasingly complex electronic equipment and miniaturization of electrical components has led to increased thermal energy densities which must be rejected to coldplates. These increased densities cannot be adequately dissipated using single-phase fluid without high liquid flow rates and the attendant system penalties. Additionally, single-phase radiators would be sensitive to micrometeoroid or space debris during the long term missions and would be difficult to replace without deservicing an entire system.

ADVANCED THERMAL TECHNOLOGY PROGRAMS

As a result of the previously mentioned limitations of single-phase thermal control technology and increasingly stringent requirements for the thermal management system being adapted for the Space Station, a more versatile and sophisticated technology is being pursued. Driving the evolution of the active thermal control system are emerging requirements for increased capabilities to accomplish the following: accommodation of changing heat loads and heat generating equipment, utilization of waste heat, moderation of equipment operating temperature affects on loop temperature, efficient rejection of waste heat to space, adaptation to system growth, transportation of thermal energy over long distances, and demonstration of long life/high reliability components.

To accomplish these requirements, increased attention is being given to heat pipe and two-phase fluid thermal transport technologies. NASA has formed the Manned Space Station Steering Committee Thermal Working Group to coordinate the efforts of the various centers and to provide a coherent plan for the attainment of specific thermal technology goals. These goals have been divided into three groups: (1) long life heat rejection, (2) versatile thermal acquisition and transport, and (3) integrated thermal utility. Table 1 gives a breakdown of the goal groups and the detailed objectives (by priority) within each group.

Long Life Heat Rejection

The long mission duration inherent in a Space Station program exposes all external areas to micrometeoroid or space debris impact. A puncture of a single-phase fluid system radiator tube would cause a loss of the working fluid and a decommissioning of a large radiator area, which would be limited only by segmentation or isolation with attendant valves and control sensors. This problem is ameliorated by using heat pipe elements to comprise the radiator panels. A puncture of a heat pipe tube would cause a loss of working fluid from only that element, allowing the remaining elements to function as normal. The radiator areas would then be much less sensitive to localized damage. Concurrently, control requirements would be simplified, pumping power reduced, and on-orbit construction and maintainability enhanced.

Grumman Aerospace Corporation (GAC), under JSC contract NAS 9-15965 has been developing a high capacity heat pipe radiator element which could be utilized in a space constructable radiator system (ref. 1). Such a concept for Space Station application is shown in figure 1. Recently completed thermal vacuum testing at JSC has demonstrated the capability of the heat pipe element to reject in excess of 2 kw. Two 15-meter U-shaped elements were tested representing module-mounted and planar radiator elements. The module-mounted type of element had heat exchangers brazed to both ends while the planar radiator type had a mechanical heat exchanger on one end. The mechanical unit utilized a thermal expansion bolt, which was

exercised during testing, pre-loaded to approximately 13,600 kg to apply pressure to the heat pipe evaporator. Preliminary test results are quite favorable.

In order to further increase the technology base for long-life heat rejection, JSC is currently preparing a Request-For-Proposal (RFP) for development of an environment sensing radiator system. Such a system would detect incident thermal energies on the deployed radiator surface and orient the radiator to minimize these energies. Significant reductions in required radiator area would result from such a mechanism. Sensitivity of the radiators to thermal coating degradation would also be markedly reduced.

Additionally, Marshall Space Flight Center has been tasked with leading the development of requirements and techniques for thermal coating maintenance and refurbishment of radiator surfaces. Results of their findings will, of course, also be applicable to thermal coatings on structure not directly related to radiators as well.

Development of advanced radiator concepts, specifically for the SP-100 project, has primarily been occurring at Lewis Research Center. Concepts which have surfaced as possible candidates for high capacity heat rejection are the liquid droplet radiator, shown in figure 2, and the liquid belt radiator, figure 3. Though these concepts offer the promise of attractively low design weights, there are some attendant problems, such as unconstrained particle impingement on spacecraft surfaces, which must be overcome.

Versatile Thermal Acquisition And Transport

As a means of supplying thermal control to a variety of electronics packages, instruments, and payloads, a centralized thermal bus concept is being developed in a contract with GAC for JSC. Ideally, this thermal bus will operate much as an electrical bus, providing a constant temperature thermal sink or heat source which can be accessed at any point in the loop. This will be accomplished by utilizing two-phase fluid technology. Mechanical pumping requirements will be small due to capillary pumping which will occur in the evaporators. Figure 4 illustrates the comparison between pumping power requirements for single-phase and two-phase thermal transport systems.

Sequencing of heat generating equipment will not be required due to the isothermality of the bus loop. Additionally, waste heat is more readily utilized, minimizing electrical heater requirements. Two-phase coldplates will also be an order of magnitude more efficient, resulting in a reduction in size and weight. Figure 5 illustrates a simplified schematic of the proposed system (ref. 2).

Goddard Space Flight Center has been leading the effort in developing high density heat acquisition and transport for the payload/experiment community using two-phase technology. After having pioneered capillary pumped and pumped two-phase loops and having accomplished a variety of flight testing of heat pipes, Goddard has

established itself as an expert in payload/experiment level thermal control. These preliminary concepts are being scaled up for vehicle level Space Station application by JSC. GSFC is currently pursuing an instrument bus thermal test bed to integrate capillary and two-phase cold plates for heat acquisition with two-phase heat transport and rejection technology. Acquisition levels on the order of 5 to 10 w/cm² are being targeted.

Within this category of versatile thermal acquisition and transport, MSFC is investigating heat transport across structural boundaries. This is specifically in development of a rotating thermal joint which is capable of repeated mating and demating operations. A proposed application would be in an articulating radiator system for solar flux avoidance. The approach being pursued is a high thermal capacity heat pipe to heat pipe rotating joint which would eliminate a requirement for fluid seals.

To establish fundamental background and framework from which the two-phase hardware is developed, reduced gravity two-phase flow basics are being pursued by LeRC, GSFC, and JSC under the Thermal Energy Management Processes (TEMP) program sponsored by OAST and the Microgravity Research Program. The TEMP 1 phase program objective is to obtain a fundamental data base necessary for the implementation of two-phase thermal management systems. Other phases of the TEMP program will be discussed later. The Microgravity Research Program complements the TEMP 1 activity in gathering fundamental data for designing two-phase thermal control components. This data gathering will conceivably take the form of a middeck flight experiment aboard the Orbiter in the late '85 to early '86 timeframe.

Integrated Thermal Utility

This goal group is directed at developing a capability for a fully integrated and automated thermal management system aboard a Space Station. Through a judicious integration of various thermally controlled subsystems, waste heat can be more readily utilized and crew involvement can be minimized. Ideally, this system will result in a significant reduction in supplemental electrical heaters and parasitic power requirements. Methods of attaining such a self-regulating system include thermal storage; heat pump augmentation; automatic system control, monitoring and fault isolation; and improved systems level analytical capabilities.

MSFC is currently addressing the thermal storage problem that is associated with both two-phase and single-phase working fluids. Such a storage capability will store peak thermal energy loads for rejection during more favorable portions of the orbit. Therefore, the radiators need not be sized to handle the total expected instantaneous or short duration peak loading, but only an orbital average heat load.

As a means of assuring adequate design of an integrated thermal system and the most favorable use of waste heat, JSC is currently sponsoring a contract with Rockwell International/Seal Beach which will conceptually and analytically develop a High Efficiency Automated Thermal (HEAT) control system. The scope of this effort is to evaluate a variety of thermal systems and components for a Space Station and to prepare a preliminary design for a promising system concept. The evaluation encompasses single and dual phase fluid thermal control systems and their interaction with thermal storage, thermal transport, radiators, and automatic system controllers. Also addressed in the contracted study will be the levels of modularity, redundancy, and maintainability which will be required for adequate long-term thermal system performance.

THERMAL TECHNOLOGY TESTING

To evaluate the analytical and theoretical basis from which thermal technology directions are determined, there have been two significant test programs which have been established. These programs will assess the components and, eventually, total systems which have been proposed for the Space Station thermal control network. The TEMP program objective will better define the fundamental data base and zero-G operation characteristics of two-phase thermal systems. The Thermal Test Bed will allow a ground-based verification of advanced thermal systems and components and will establish the data base necessary to confidently commit advanced thermal technology for implementation in the Space Station flight development program.

TEMP Program

The TEMP program is composed of four interactive phases. TEMP 1, primarily investigated by LeRC, will measure the basic characteristics of two-phase fluid behavior in microgravity conditions. Experiment objectives will be the determination of two-phase flow regime boundaries, evaluation of heat transfer characteristics in two-phase flows, and assessment of flow boiling pressure drop. These phenomena will be investigated in flight experiments aboard the Shuttle located in a middeck locker area.

Testing of specific components which will exist in a two-phase thermal management system will take place in the TEMP 2 phase managed by GSFC. Representative evaporators, condensers, capillary pumps, and heat pipes will either be mounted inside a GetAway Special (GAS) container in the payload bay of the Shuttle or on a payload carrying structure (DFI pallet type). Operating parameters of the components will be determined and design characteristics established for incorporation into the ultimate Space Station thermal control network.

TEMP 3, investigated by JSC, will look at the system-level aspects for more large scale components of the thermal control system to be flown aboard a Space Station. Initially, a full scale element, 15 meters long, of the space constructable radiator will be tested as it

is mounted to the Orbiter payload bay sill longeron. A subsequent flight will test the radiator/heat exchanger assembly and the use of the Orbiter RMS (Remote Manipulator System) to make and break the connection between the two. The next step, TEMP 3C, will be a flight test of a representative portion of the thermal management system including thermal bus, hot plates, cold plates, radiators, controls, and associated pumps. The flight of this system, projected for 1987, will give valuable information on the actual operation limitations and control requirements of a large scale, two-phase thermal management system.

The fourth phase of the TEMP program, TEMP 4, is being investigated by MSFC. This portion of the program will evaluate thermal control coatings for thermally sensitive surfaces and will assess techniques for coating refurbishment or replacement on orbit.

Results of the TEMP program promise to give design engineers much needed information on proposed thermal management techniques for a Space Station. When this information is coupled with that extracted from the Thermal Test Bed, a coherent and well-tuned thermal management system should be possible.

Thermal Test Bed

Johnson Space Center was recently announced as the lead center for development of the Space Station Thermal Test Bed (TTB). This test bed, which is a cooperative effort with GSFC, LeRC, and MSFC, will be used for testing and evaluation of thermal technology alternatives and system configurations for incorporation into a Space Station. Hardware trades will be made to verify the applicability of various heat acquisition, transport, and rejection methods. The Test Bed will conceptually evolve from a single-phase fluid system with some two-phase components into a fully two-phase system. This evolution should aid in quantifying performance and development risk for new approaches to thermal control. More specifically, it will provide the "transfer function" for new two-phase/heat pipe technology into the Space Station development program.

As the Test Bed becomes more mature it will be used for establishing automated control requirements, fault detection and isolation methods, and on-orbit maintenance and refurbishment compatibilities. It will also enable a more realistic evaluation of desired thermal integration levels and could provide some otherwise unforeseen opportunities for thermal system enhancement. Table 2 gives an indication of the expected Test Bed end products which will be in support of the Space Station Initial Operational Capability (IOC). These products have been established along the same lines as the Space Station Steering Committee Thermal Working Group technology goals for consistency with the overall program direction.

Preliminary results of Test Bed experimentation will feed directly into the TEMP program and vice versa. Components for flight testing in TEMP will go through cursory testing in the ground-based Thermal Test Bed to verify performance expectations during microgravity conditions.

Likewise, initial results from the flight tests will be correlated with ground-testing data and more extensive test requirements will be formulated. Conceptually, the TEMP 3C flight test article will be a direct outgrowth of the inter-center Thermal Test Bed.

SPACE POWER SYSTEM THERMAL REQUIREMENTS

The accommodation of wide ranges of thermal loads and changing thermal control requirements is a specific objective of developing two-phase thermal technology. As such, space power system thermal requirements should be readily attainable in the near-term (within the decade). However, there are some developing areas which will require greater attention for their integration into the thermal management system scheme.

"Black-box" electronics for power conditioning and distribution will be primary users of the Space Station thermal control system. Their requirements for thermal conditioning are currently being assessed for either an evolutionary single-phase or developmental two-phase system. The temperature ranges which these components are expected to operate within (approximately 0° to 80°C) are within the scope of state-of-the-art systems and those being developed in the two-phase technology area. It is necessary now that the specific operational requirements of this equipment be made available to the thermal designers so that their temperature needs can be incorporated into the integrated Thermal Management System.

As it is perceived at this stage of planning, the TMS of a Space Station will be segmented into two, and possibly three, independent temperature loops. The two most probable control temperatures will be 20°C (70°F) and 80°C (180°F). These loops will control general housekeeping/equipment loads and fuel/regenerable fuel cell loads, respectively. The third possible loop, if refrigeration is not used, would be at approximately 4°C (40°F), taking care of environmental control and payload/experiment thermal needs.

In all of the proposed scenarios, the power generation equipment will have a dedicated thermal loop due to the perceived higher operating temperatures of the hardware. (Nickel-cadmium batteries would, of course, not be included in such a system due to their low temperature requirements.) The technologies of both single and two-phase systems will be stretched if control temperatures rise much higher than 95°C (200°F), however. It is at this point that a new technology direction would need to be defined and incorporated into the thermal technology development program.

Two-phase working fluids for thermal control of high temperature systems become almost mandatory due to the high heat flux densities and total thermal load to be transported. Though high temperature heat pipes have been developed for the nuclear industry, their use in a space environment may be constrained by material considerations (i.e., weight, durability, compatibility). Furthermore, heat transport distances in a Space Station application could be considerably longer

than in terrestrial applications. The fundamental limitations of current two-phase and heat pipe technology in high temperature energy transport must be determined and new developmental directions highlighted if thermal control capabilities are to keep pace with thermal energy generation growth.

CRITICAL TECHNOLOGY BARRIERS

Though a large analytical and experimental data base supports the use of two-phase fluid technology for thermal control of a Space Station, hardware flight testing has been limited to heat pipe concepts. These flight experiments, flown aboard the Orbiter vehicle and various satellites, have indicated that heat pipes are a viable and extremely useful technology in the low gravity fields of space environs. Heat pipes are, in fact, best tested in the space environment, away from gravitational influences which tend to degrade their performance.

Two-phase thermal transport loops, however, have not yet been flight tested. Small scale ground testing has demonstrated some promising results for this technology, which is a justification for pursuing it further, but there has not yet been space testing which would substantiate these results. This will be occurring within the next two to three years within the framework of the previously described TEMP program. The real question becomes whether this technology will be suitable to such large scale systems as are proposed for the space station. Can the system be readily grown along with the Station and, if so, what are its ultimate limits in temperature range, heat acquisition, transport, and heat rejection? Future flight testing should make progress in the evaluation of this two-phase technology for use on space vehicles, but the results of these tests will determine the extent to which this technology is pursued further.

Within the two-phase technology verification effort is the requirement to understand transient effects on the operation of the thermal system components and on the system as a whole. Uncontrolled transient heat fluxes could potentially choke the system, causing a blockage of mass and/or heat transport. Additionally, cold start-up and large turn-down ratios of the thermal system present unique problems. This is more significant in high temperature two-phase systems due to the fact that there would actually be three phases (solid, liquid, and vapor) present during a cold start/high turn-down condition.

Also in the high temperature thermal control technology area is the possible requirement to transport heat over long distances. The actual users of high energy heat have not yet been well defined, so this requirement is not strong. However, the production of high temperatures for use in material processes and experiments may be one of the potential justifications for more sophisticated energy conversion techniques. These requirements must be more well delineated to understand what the capacity of future thermal systems must be.

PROPOSED TECHNOLOGY PROGRAM

In addition to the thermal technology program directions which have been previously outlined, it is proposed that additional high temperature goals be charted for incorporation or for parallel investigation, with the existing program. These goals would include the development of high temperature thermal transport (thermal bus) concepts and complementary hardware such as heat pipes, hot plates, and heat exchangers.

Another technology area which would be related to high temperature technology would be vapor-compression. This would permit lower temperature vapors to be elevated in temperature in order to utilize existing high temperature radiators for more efficient heat rejection. Previous evaluations of utilization of this technology have shown that it would not be cost effective, but more efficient power generation methods could cause vapor-compression to become more attractive.

A preliminary technology timetable and funding level is illustrated in figure 6.

CONCLUSIONS

NASA is diligently pursuing an ambitious program to develop advanced two-phase/heat pipe thermal technology for use in large space vehicles with the Space Station as an early focus. The advantages of such technology over single-phase thermal control are long life heat rejection, versatile thermal acquisition and transport, and an integrated thermal utility. Hardware which is developed out of this technology program will eventually be incorporated and evaluated in the JSC managed Thermal Test Bed. The Test Bed will serve as a focal point for significant technology to be utilized on the Space Station and will aid in establishing operational limitations and control requirements.

Orbital flight testing aboard the Orbiter vehicle of significant thermal technology components and test sections will be accomplished under the TEMP program. Ground-based testing results and the proposed Space Station thermal control system will be verified on orbit so that the final system design can be approached with confidence.

State-of-the-art space power system components should integrate well into two-phase thermal bus concept. However, more advanced concepts, such as solar dynamic and nuclear power systems, will require significantly more thermal technology development, particularly in the higher temperature ranges. Program directions must be determined and program goals established in order to coherently approach the problem of thermal control for developing technologies.

There are no apparently insurmountable technology barriers confronting advanced thermal control development. It must be emphasized, however, that much is yet unknown about two-phase phenomena

in a microgravity environment. Knowledge gaps will be rapidly filled within the next few years due to the technology program currently in place. A similar program will be required for high temperature two-phase technology to evolve and mature.

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TABLE 1

THERMAL TECHNOLOGY GOALS AND OBJECTIVES (PRIORITY ORDER)

GOAL	OBJECTIVES
Long Life Heat Rejection	<ol style="list-style-type: none"> 1. High Capacity Heat Pipe Radiator 2. Deployable/Constructable Radiator System 3. Environment Sensing Radiator System 4. Body Mounted Radiators 5. Thermal Coating Maintenance/Refurbishment 6. Advanced Radiator Concepts
Versatile Thermal Acquisition and Transport	<ol style="list-style-type: none"> 1. Centralized Thermal Bus Transport 2. High Density Heat Acquisition 3. Heat Transfer Across Structural Boundaries 4. Reduced Gravity Two-Phase Flow Basics
Integrated Thermal Utility	<ol style="list-style-type: none"> 1. Thermal Storage/Load Leveling/Refrigeration 2. Automatic System Control/Monitoring/Fault Isolation 3. System Integration Analysis/Trades 4. Thermal Computer Model Improvement 5. Ground Test Capability 6. Inflight Handling and Maintenance

TABLE 2

**THERMAL TEST BED END PRODUCTS
SUPPORTING IOC**

1. Environmentally insensitive, constructable heat rejection system
 - a. High capacity heat pipe (includes flight experiment verification)
 - b. Body mounted radiator
 - c. Space constructable radiator elements (includes flight experiment verification)
 - d. Gimbaledd radiator subsystem
 - e. High conductivity radiator fin
 - f. Thermal coating replacement/refurbishment techniques (includes flight experiment verification)
2. Versatile thermal acquisition & transport for multi-disciplinary users
 - a. Two phase coldplates (includes flight experiment verification)
 - b. Two phase heat transport bus (includes flight experiment verification)
 - c. Contact heat exchangers
 - d. Instrument thermal bus (includes flight experiment verification)
 - e. Long life fluid system components
3. Integrated thermal utility system
 - a. Thermal storage device
 - b. System automated control techniques
 - c. Fault detection/isolation techniques
 - d. On-orbit maintainability techniques
 - e. Ground test techniques for two-phase systems

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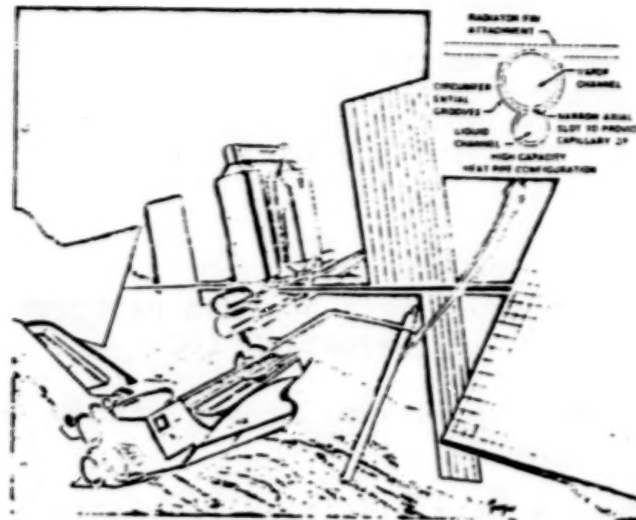


Figure 1. - Space constructable heat pipe radiator.

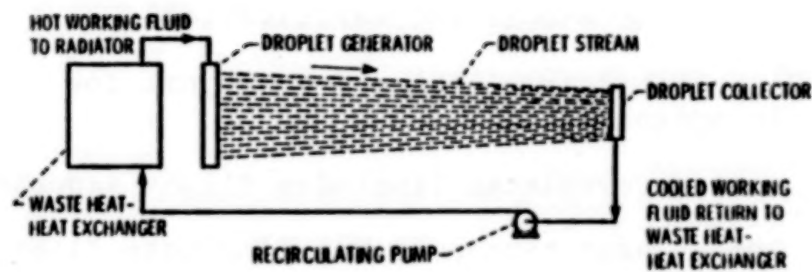


Figure 2. - Liquid droplet radiator concept.

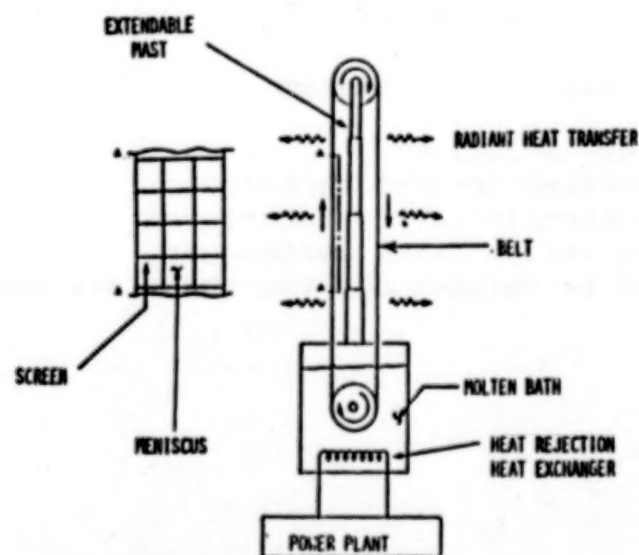


Figure 3. - Belt radiator.

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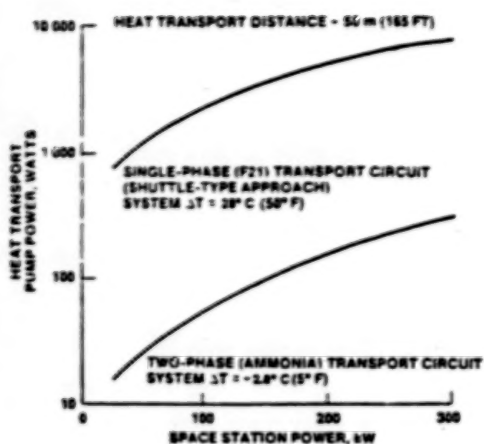


Figure 4. - Comparison of single- versus two-phase heat transport circuit power requirements.

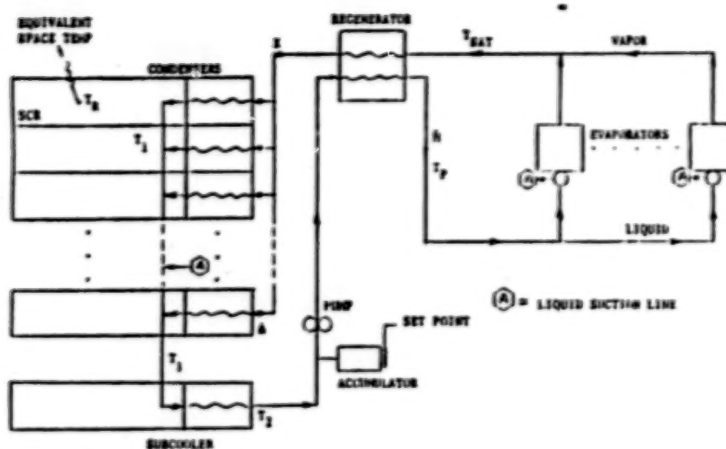


Figure 5. - Bus preliminary design schematic.

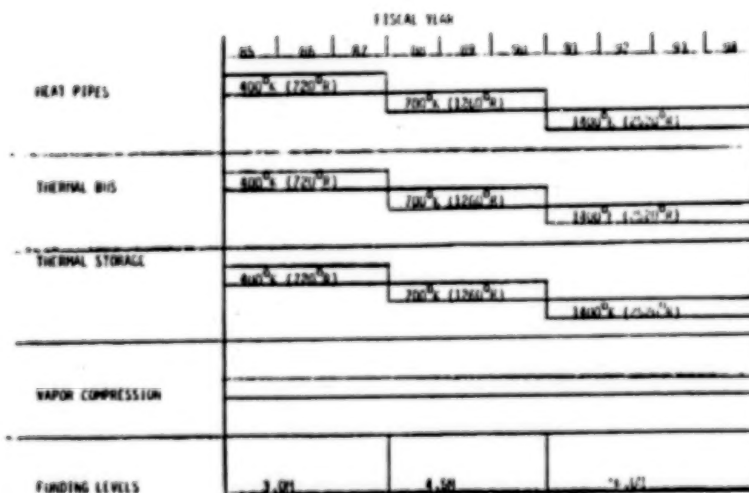


Figure 6. - Proposed high temperature two-phase thermal technology development program.

SPACE POWER MANAGEMENT AND DISTRIBUTION STATUS AND TRENDS

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An overview of space power management and distribution (PMAD) is provided which encompasses historical and current technology trends. PMAD components discussed include power source control, energy storage control and load power processing electronic equipment. Status of distribution equipment comprised of rotary joints and power switchgear is evaluated based on power level trends in the public, military and commercial sectors. Component level technology thrusts, as driven by perceived system level trends, are compared to technology status of piece-parts such as power semiconductors, capacitors and magnetics to determine critical barriers.

INTRODUCTION

Future power management and distribution (PMAD) system design requirements as defined by public, military and commercial advanced design organizations are very challenging. PMAD designers stand at the threshold of a new era looking to a future where low voltage power systems technology is no longer appropriate. Component technology cost, efficiency and weight trade-offs point toward design solutions which require higher voltage and frequency. This paper addresses PMAD technology issues and provides recommendations for further evaluation of technology and programmatic direction by workshop working groups.

REQUIREMENTS SUMMARY

Near and mid-term power requirements in low earth and geosynchronous orbit will be met, in large part, by photovoltaic/battery systems. Radioisotope thermoelectric generators and, possibly, radioisotope dynamic power sources may be used for missions with unique requirements. Mid-term power in low earth orbit is driven by space platform and space station missions having initial power requirements in the range of 15 to 100 kW. Commercial satellites will be dominated by geosynchronous communications applications in the 2 to 20 kW range. It is assumed for this analysis that other commercial ventures such as manufacturing in space are enveloped by space station (public) power projections. In the far term, the SP-100 nuclear reactor will provide another option at the 100 kW power level.

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Projected NASA, military and commercial space power requirements for near and far term are presented in reference 1-6. From these documents and other sources, one can broadly summarize power level trends as follows:

Load Power Level (kW)			
Sector	1985-1990 (Near Term)	1990-1995 (Mid Term)	Post 1995 (Far Term)
Military	2 to 12	12 to 30	30 to 100
Military (Pulse Loads)	—	50 to 250	>1000
Public (NASA)	2 to 15	2 to 100	100 to 250
Commercial	2 to 4	2 to 8	>10 to 20*

*Assumes Payload weight limit, 4800 kg, geosynchronous orbit 20% weight allocation to power subsystem, 20 W/kg.

In practice, high power requirements may be met by combining the outputs of a number of power channels, each consisting of power sources, energy storage and PMAD components. This is a logical continuation of the Skylab approach where 18 power channels were used for the ATM and 8 power channels comprised the Airlock Module power systems. Recent studies employing a parallel channel approach are discussed in reference 7 and 8.

Typical channel size ranges from 3 to 18 kW, depending on total system load capability. It is important to keep channel power requirements in mind when evaluating PMAD component ability to satisfy future requirements. Channelized systems impose less stringent design requirements than systems with centralized power processing which have higher PMAD component power throughput and thermal loading. With large nuclear reactor power sources, channelization may be neither practical nor desirable. This may impose significantly higher power requirements on individual PMAD components in the post 1995 time frame.

PMAD SCOPE

As viewed by NASA, the scope of PMAD has broadened with time to include related technologies such as flywheel energy storage, environmental interaction, laser power transmission, analytical modeling, autonomy and thermal management (ref. 2). Several of these technologies are the main topic of other papers in this document. This paper focuses on PMAD equipment within the dashed lines of figure 1, which consists of power electronics and distribution components.

POWER SOURCE CONTROLS

Future power sources (whether they be solar arrays, nuclear, solar thermal, or other devices) will require some form of control to properly regulate power output and ensure efficient power source operation with minimum degradation. The historical trend in spacecraft power source controls has been to accommodate increasingly powerful solar array and nuclear sources while maintaining power electronics heat dissipation

within the capability of spacecraft passive thermal control techniques. Although power source controllers are becoming more efficient, power throughput and subsequent heat dissipation is increasing beyond the capability of passive thermal control methods. Active thermal control consisting of heat pipe and/or fluid loop cooled mounting panels (as on Skylab) will be used more frequently to keep power components within acceptable temperature range.

Figure 2 shows solar array control component power capability trends (ref. 9-16). Power control capability has increased by more than an order of magnitude from the late 1960s to the present. Near and mid-term launch vehicle payload weight constraints will limit high power satellites to low earth orbit missions. Low earth orbit load power capabilities of the programmable power processor (P³), and solar array switching unit (SASU) are compatible with multi-channel PMAD systems concepts requiring approximately 5 kW per channel. Large 200 kW low earth orbit systems will require approximately 40 power channels each having power source, energy storage and power distribution components if technology remains frozen at present levels. This proliferation of equipment would not be cost effective. A more reasonable approach would be to develop power source controllers capable of handling channelized loads of 10 to 20 kW (source power of 20 to 40 kW). The P³ is a microprocessor controlled series switching regulator developed by MSFC. As a solar array controller, it functions as a maximum power tracking buck regulator. The SASU is a digital solar array control method which uses semiconductor switches to connect or disconnect array sections as required for load support while limiting the bus voltage to a predetermined value.

Significant differences exist between the P³ and SASU. The P³ array maximum power point tracking capability provides better array power utilization during periods when the array may be off-pointed from the sunline by large angles and at beginning-of-life. On the other hand, the SASU has lower dissipation for a given power level as shown in figure 3.

Power source control efficiency has been continuously improved such that dissipation can be expected to be less than five percent of controlled array power output. The shunt switch SASU dissipation is less than one percent of array output. This translates into 1 to 5 kW dissipation for a 100 kW system, depending on the type of solar array control selected.

Another significant difference between the P³ and SASU is the allowable array operating voltage. For low earth orbit, P³ load bus voltage may be restricted to the 100 to 150V range since the P³ allows the array voltage to double at eclipse exit. The critical 300 to 500V region must be avoided where undesirable plasma interaction may occur causing array power loss and arcing. The shunt-connected SASU allows higher load bus voltages (200 to 300V) since the array is directly connected to the bus (eclipse exit voltage is limited to the bus voltage by the SASU).

Full shunt and SASU control methods work equally well for solar arrays and thermoelectric sources which require constant voltage operation. Technology is presently available to accommodate near and mid-term power levels. In the far term, pulse loads, higher power levels, launch vehicle weight limitations, and dynamic power sources will require advancements in piece-part component technology to improve efficiency, reduce weight and increase heat transfer capability.

POWER PROCESSORS

Power processors consist of regulators, converters, inverters and battery chargers. Present trends are to incorporate role adaptability using embedded microprocessors with software programmable operating modes. Thus, the versatile P3 (ref. 12) may be used as a power source controller, battery charger or line regulator.

Power processor design status and trends are shown in figure 4. Data are from references 12, 13, and 16 to 21. In the mid-1970s significant breakouts from the shaded lower left hand corner occurred. Ion engine power processor technology developments and the Canadian Communications Technology Satellite (CTS) power processor broke new ground. High power high voltage SCRs and transistors, heat pipe cooled magnetics, improved capacitors and power diodes developed under sponsorship of LeRC led to successful and rapid power processor technology advancement. The rate of technology development was enhanced by close interaction of power processor designers and advanced piece-part component vendors such that component application problems were surfaced and resolved in a timely manner. This cooperation should be continued in future component development programs.

Since 1974, LeRC and MSFC have sponsored development of power processors in the 10 to 25 kW range having input voltage capability compatible with 200 to 400 Vdc system power buses. The most advanced power processor is the 12.5 kW P3 which has reached the prototype stage.

The Space Shuttle Orbiter Power Extension Package (PEP) program spawned two 6 kW transformer coupled converter power processor developments capable of providing PEP/Orbiter power ground isolation. Under Air Force sponsorship, 10 kW and 200 kW Schwartz converters capable of 600V input are being breadboarded with additional work underway by Martin Marietta to hybridize their control circuits. Reference 19 is an Air Force study of airborne and spaceborne power processors having 250 and 500 kW continuous outputs.

Component Developments

With the advent of high power space missions, power conditioning equipment will be required to have higher power capability per unit weight. In order to accommodate this design requirement, development program thrust is aimed at increasing input bus voltage and operating frequencies while maintaining or improving efficiency.

High frequency ac power distribution systems using resonant power conditioners are also being developed to reduce weight. Resonant power circuits use sinewave currents which minimize turn-on and turn-off losses with inherent zero current switching (ref. 22). Relatively slow, high power components can be operated above 10 kHz with minimal efficiency penalty.

Power conditioning equipment design is constrained by high power component technology limitations. Table 1 lists four basic high frequency power components which are key to implementing ac or dc PMAD systems. Efficient, reliable power supplies for high power transmitters, electric propulsion, laser communications, science instruments and spacecraft housekeeping can only be realized through continued improvement in these components.

High power components required for pulse power (energy compression) applications are shown. Part classification is listed along with peak ratings, performance trends and critical component barriers. Design operating power with voltage and current derating is about 30% of peak power rating.

NASA Lewis Research Center has sponsored most of this high power development except for the pulse power components (ref. 23 to 27). High power component work is progressing satisfactorily in all areas except for FETs and dc capacitors. Power FET technology improvement should be relatively straight forward because of extensive lower power component development work in progress. DC filter capacitor development is not funded at present. To obtain high capacitance required for high power applications, many paralleled components are used.

Present ratings are adequate for travelling wave tube amplifier (TWTA) applications. Direct broadcast satellite TWTA electrical power conditioners are approaching 1 kW. Additional component work will be needed in this area as payload RF output power and frequency increase, causing power conditioner output voltage requirements to exceed available component ratings. Semiconductor, capacitor and magnetic device state-of-the-art is summarized in figures 5 and 6.

Critical component barriers limit the practical operating frequency due to switching and recovery loss characteristics. Component insulation barriers limit operating voltage. Heat dissipation associated with power loss and voltage stress are important component reliability factors.

For future high power equipment, extensive work is required to establish component specifications, qualify components for flight programs and gather component reliability data. Careful selection of components for development is essential to minimize costs and meet program schedule requirements.

POWER DISTRIBUTION

Several studies have examined power distribution bus voltage and waveform as applied to future high power space missions. References 3, 4, 7, 8, and 28 have supported ac, dc or hybrid ac and dc systems with strong arguments favoring their choices. Prior studies have been hampered by a paucity of objective data. Critical information such as rotary power transfer device performance at high voltage, subsystem level test data, and detailed payload power supply interface definition is not now available.

There appears to be general agreement that future large space power systems will employ both ac and dc. The question is, where in the PMAD will ac and dc be used? We recommend resisting the temptation to answer this question until test results are available from NASA large space power system ac and dc test beds allowing an objective decision to be made. The recommendation of reference 29 to delay this choice until completion of space station phase B trade studies seems entirely appropriate at this time. In the interim, parallel development of both ac and dc components should proceed.

To date, public, military and commercial spacecraft bus voltage has been in the range of 20 to 50 Vdc with rare exceptions. Satellites employing operation of payloads only in sunlight, such as the Canadian CTS and, more recently, 4 kW class domestic direct broadcast TV satellites have selected higher voltage buses in the range of 76 to 100 Vdc for input to travelling wave tube power processors. The Solar Electric

Propulsion study selected 200 to 400 Vdc for input to 25 kW ion propulsion power processors. Power source voltage in the range of 150 to 300 Vdc is being considered for 100 kW class low earth orbit platforms and stations. Operation at the upper end of this range captures most of the cost, weight and efficiency advantages of higher bus voltages while staying below the 300 to 500V region where undesirable environmental interaction may occur (ref. 3, 5, 7, 30). Well insulated power bus networks not exposed to the space environment may be capable of higher voltage transmission especially if ac distribution is used. Distribution equipment component state-of-the-art is summarized in figure 7.

Rotary Power Transfer Devices

Power transfer across rotary joints has been successfully accomplished in the past using slip ring assemblies, cable wraps, and twist-flex devices. Spacecraft slip ring assemblies on spinning/despun gyrostatt satellites have successfully operated for more than eight years at 60 rpm. At low voltage (20-50V) and reasonably high current (50 to 70 amps), slip rings are reliable and relatively noise-free. Slip ring assemblies are not environmentally sealed which raises arcing concerns if used in higher voltage (>150V) applications. The authors are unaware of any tests performed on satellite slip ring assemblies at voltages of 200 to 300 Vdc. Tests are required to eliminate uncertainty in this area and should be given high priority.

Rotary power transformers have been built and tested by LeRC and JPL (ref. 31) and a 3 kW unit is under development for the Air Force Talon Gold program. A 100 kW unit comprised of four 25 kW transformers has been studied by General Electric (ref. 32).

Power Switchgear

Electromechanical relays cannot be used at higher voltages unless their contacts are protected by arc suppression electronics. As an alternative to this cumbersome approach, LeRC has developed high voltage remote power controllers (RPCs) as discussed in reference 33 to 36.

The high power semiconductors discussed in previous sections have enabled development of a family of RPCs for high voltage dc applications. RPCs with ratings of 1 to 80 amps have been developed by Westinghouse and LeRC using bipolar and MOSFET devices. A new family of 20 kHz, 440 Vac devices is under development.

Very high power 20 to 50 kW RPCs have been breadboarded by LeRC for ion engine applications. These units use GTO thyristors, bipolar transistors and parallel/series MOSFETs to extend power and voltage ranges beyond 8 kW and 800 Vdc. RPCs may require high voltage fuses at their input for bus protection against RPC internal faults. These fuses require development.

Conductors

Conductor options include copper, aluminum, copper clad aluminum, and intercalated graphite. Copper, aluminum and copper clad aluminum have all been flight proven. Flat aluminum conductors were used to minimize power source cabling weight on Pioneer 10/11 spacecraft. Copper clad aluminum wiring has been used for weight reduction on commercial communications spacecraft (RCA) and as braided shield material on the Viking

interplanetary spacecraft. Copper clad aluminum wiring provides more reliable terminations than aluminum.

Intercalated graphite conductors are being studied for possible future use and little empirical data exists at present. If successfully developed, intercalated graphite could provide lighter weight spacecraft harnesses.

The inductance of existing wire harnesses would result in unacceptable voltage losses at frequencies in the vicinity of 20 kHz which are currently being considered for advanced systems. Coaxial power conductors are under development to provide low inductance wiring for high frequency ac distribution systems.

CONCLUDING REMARKS

Power management and distribution is an enabling technology for post 1995 space missions. Advanced dc power processors exist in the 6 kW load range. Future missions at the 250 kW level will require higher power components in the 20 to 40 kW range for cost effectiveness. Piece-part component technology developments compatible with high voltage, current and heat dissipation will be needed.

While much of the work associated with dc component development will be applicable to ac systems, a parallel effort to build and test ac inverters, four quadrant converters, rotary transformers and RPCs should be pursued. Piece-part component development effort associated with capacitors and magnetics vital to high power resonant inverter technology development must be extended.

Ac and dc test bed evaluations should be continued in parallel for the foreseeable future to enable system design choices to be made based on objective test data.

Military spacecraft steady-state and pulse power levels projected for the near term are enveloped by NASA space station class technology development. In the far term (post 1995) pulse and burst power levels (>1MW) exceed the capability of planned near-term technology developments.

The rapid development of direct broadcast satellite technology has caused transmitter power processor requirements to increase by an order of magnitude into the 0.5 to 1 kW area. In the next decade, new applications could result in another order of magnitude leap. Higher voltage and power levels will require advanced PMAD component technology development.

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TABLE 1: PMAD Piece-Part Peak Rating, Trends and Barriers

COMPONENT	PEAK RATING	PERFORMANCE TRENDS	CRITICAL COMPONENT CHARACTERISTIC BARRIERS
SEMICONDUCTORS			
Bipolar Transistors			
• Westinghouse 2T57 (2T57W12)	500V, 125A $t_s = 4\mu s$	higher frequency operation >30kHz	Turn-on, turn-off, storage time
• Westinghouse 2M57 (2M57W12)	600V, 40A $t_s = 4\mu s$		
Thyristors			
• Westinghouse T71H (T71H12W12)	1200V, 400A $t_s = 10\mu s$	higher frequency operation	Long turn-off times (t_{off})
• Westcode R220A (R220AH12F30)	1200V, 400A $t_s = 25\mu s$		
IGBTs			
• Reserco RT459	1200V, 10A	higher current/higher frequency	Long turn-off time and low power
• Matsushita 10PT200C12	1200V, 200A $t_{off} = 8.3\mu s$		
• Westinghouse 40M5	1200V, 200A $t_{off} = 10.0\mu s$		
IGT			
• International Rectifier (IRF457)	500V, 13A	higher current/higher voltage	Gate capacitance (drive losses)
• Siemens (80F54)	1000V, 4.7A		
Reverse Blocking Diode Thyristor			
• Westinghouse 162H (162H10130300) Pulse power application	1000V, 125A Pulse Current Current 2500A $t_s = 5\mu s$	higher current/higher voltage	High turn-on losses. Breakdown voltage 8-10
RECTIFIERS			
Power Diodes			
• Westinghouse P502 (P5021210R340)	1200V, 100A $t_{rr} = 300ns$	higher frequency	Long recovery time, high losses.
• Westinghouse H522 (H5221210P100)	1200V, 300A $t_{rr} = 500ns$		
High Voltage Diodes			
• Solid State Devices (SPDM)	1000V, 1A $t_{rr} = 250ns$ per element	higher frequency/higher voltage	Long recovery time, high losses.
CAPACITORS			
AC			
Polypropylene Component Research	1000V, 100A 100kVA	Higher VA rating	Hermeticity; current carrying seal; current rating; dielectric
DC Low Volt (<100V)			
Tantalum	3.6 joules	Increased capacitance per volume.	Dielectric
DC Med Volt (1-600V)			
PV2 Component Research	600V, 30uF 3.6 joules	Increased capacitance per volume	Dielectric
DC High Volt (>200V)			
Mica Custom Electronics	25kV, .01uF 2 joules	Increased capacitance per volume	Dielectric
DC Pulse			
Maxwell	25kV, 0.3uF 94 joules	10^5 pulses	Number of pulses; Dielectric losses; Current rating; Pulse current losses; di/dt
MAGNETICS			
Inductors			
	30.0kVA (1)	Higher VA rating higher frequency	Core loss (Resonant); Thermal transfer
Power Transformers			
	25.0kVA (1)	Higher voltage, higher power, higher frequency	High core loss, thermal transfer, high current, high frequency losses, high voltage insulation.

NOTES: (1) Operational value.

t_{rr} = Diode Reverse Recovery Time
 t_s = Storage Time
 t_{off} = Turn-off Time
 t_q = Turn-off Time

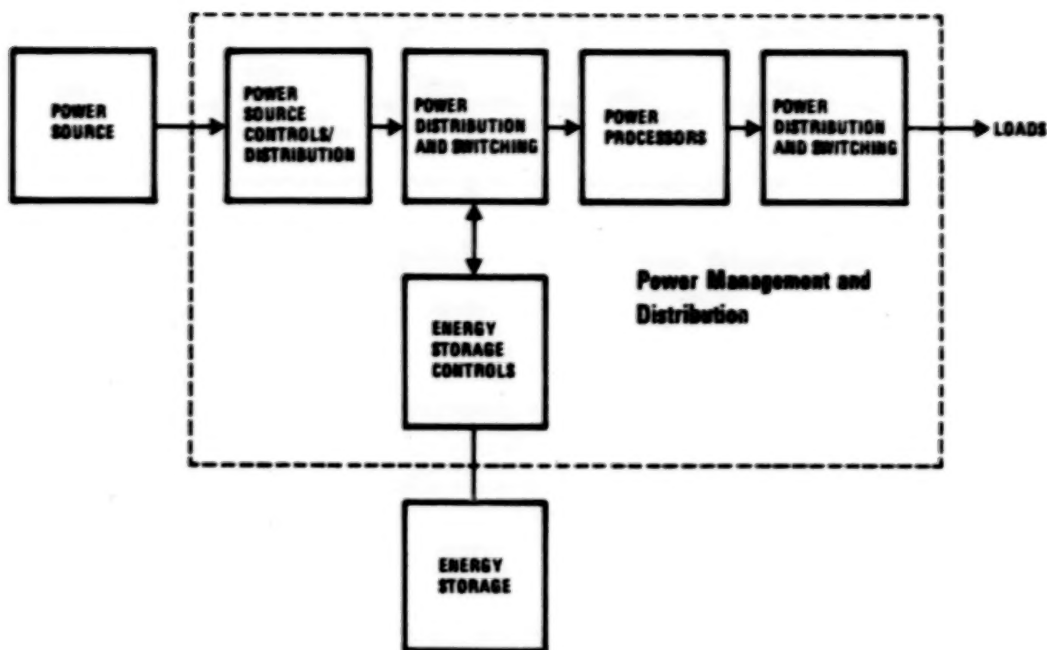


Figure 1. - Power management and distribution.

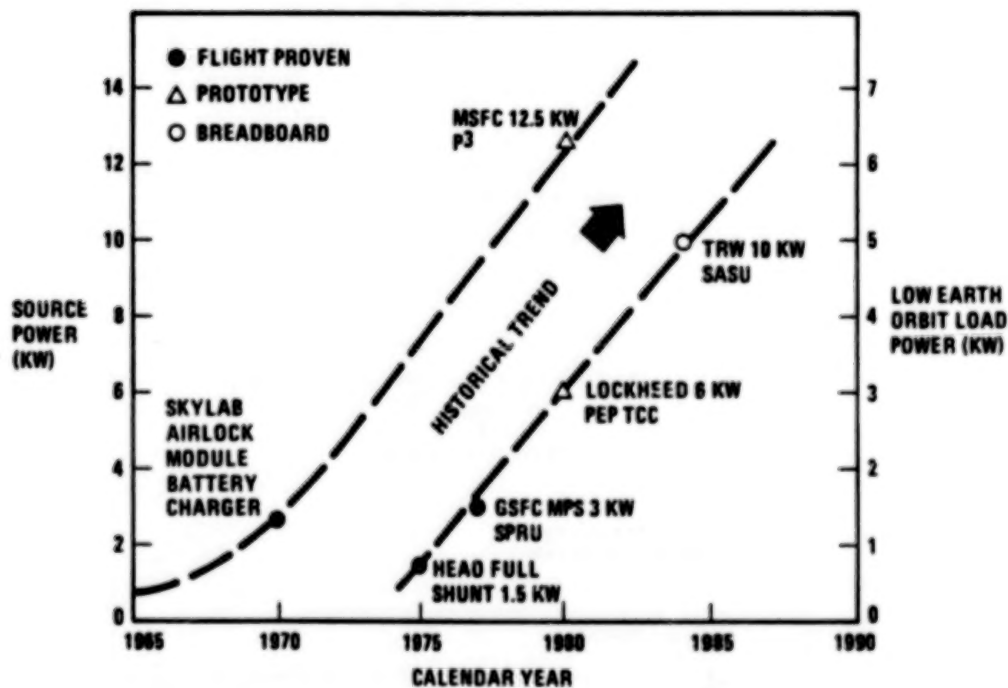


Figure 2. - Power source control capability.

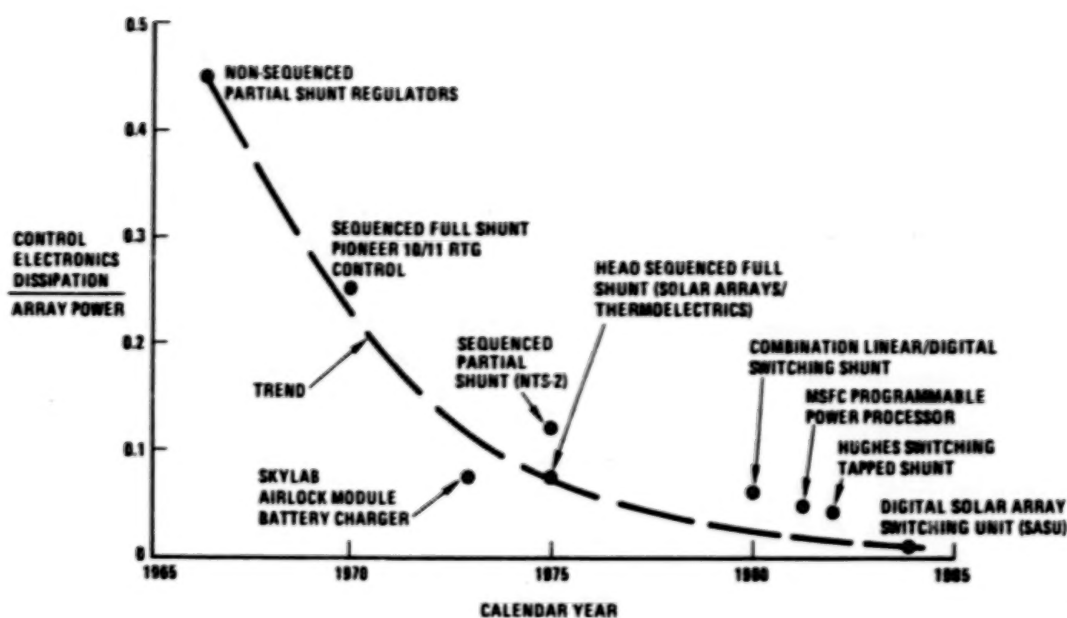


Figure 3. - Power source control efficiency/heat dissipation.

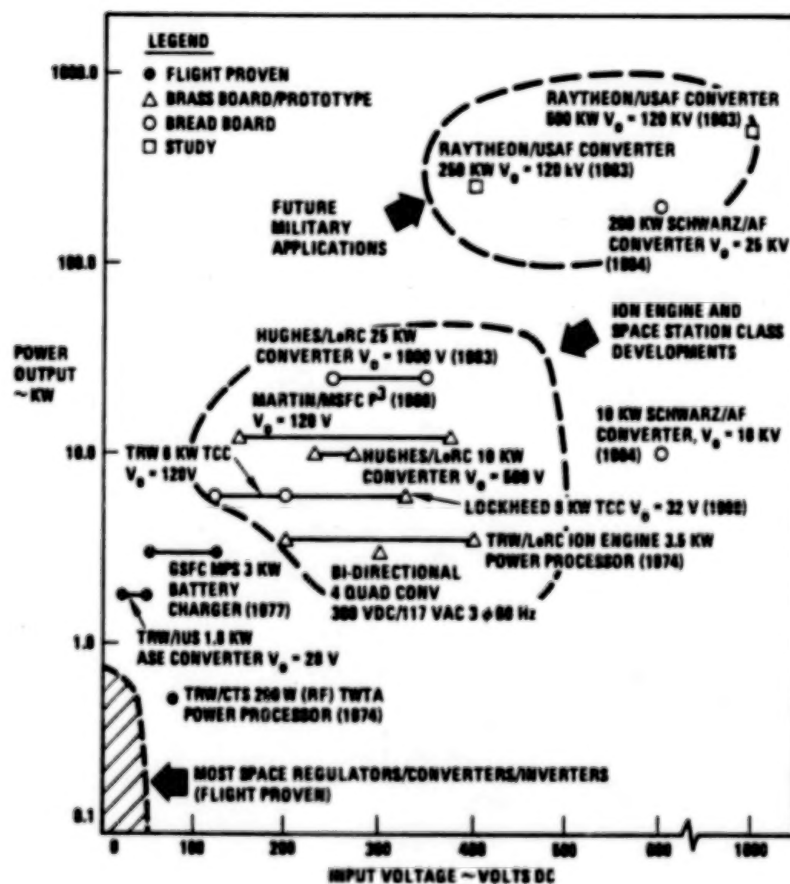


Figure 4. - Space power processor design status and trends.

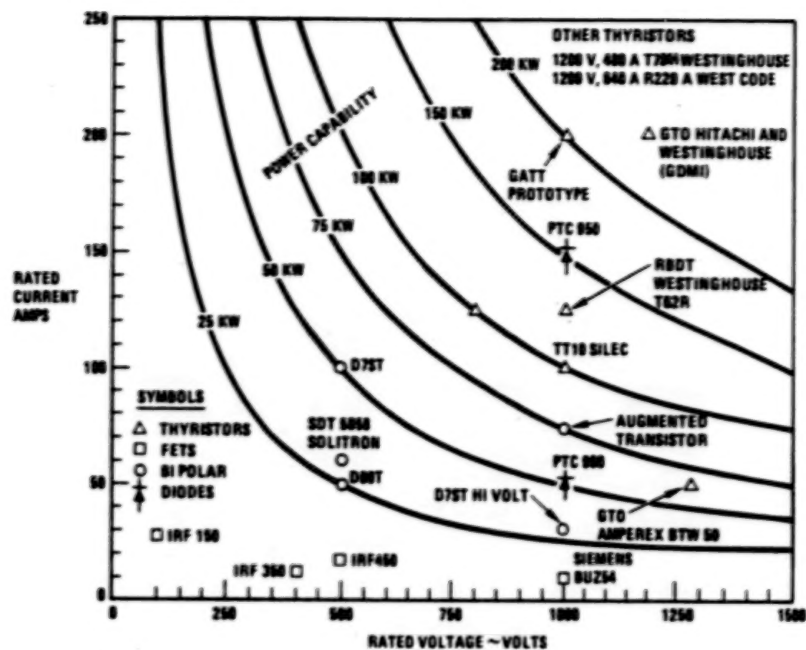


Figure 5. - Semiconductor devices.

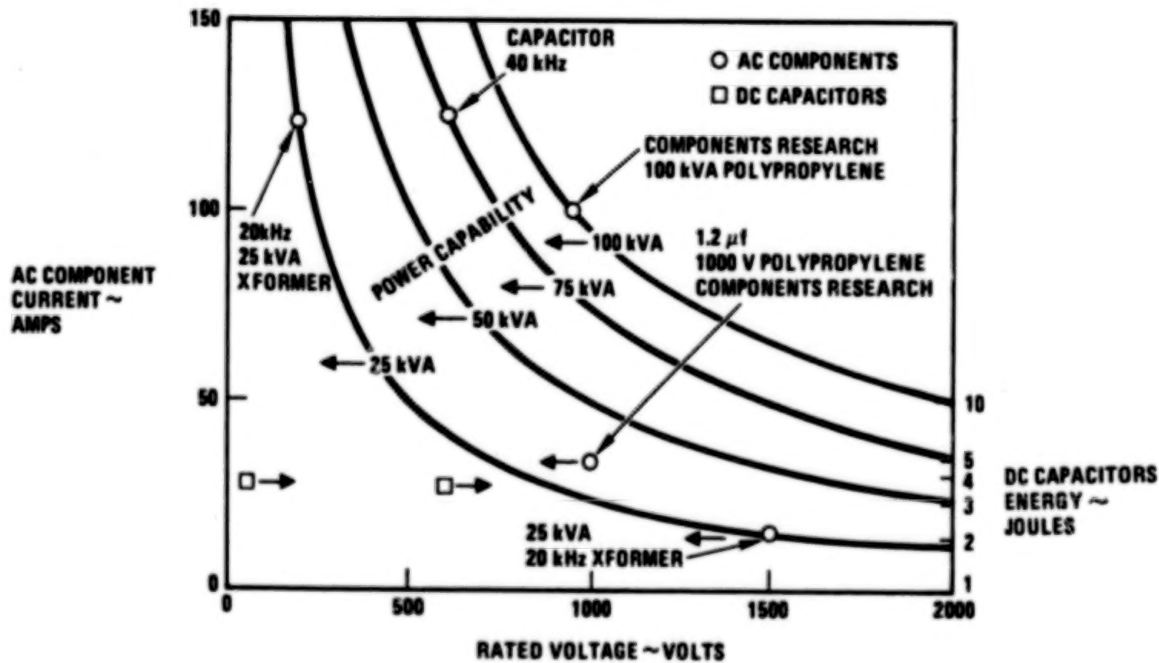


Figure 6. - Capacitor and magnetic devices.

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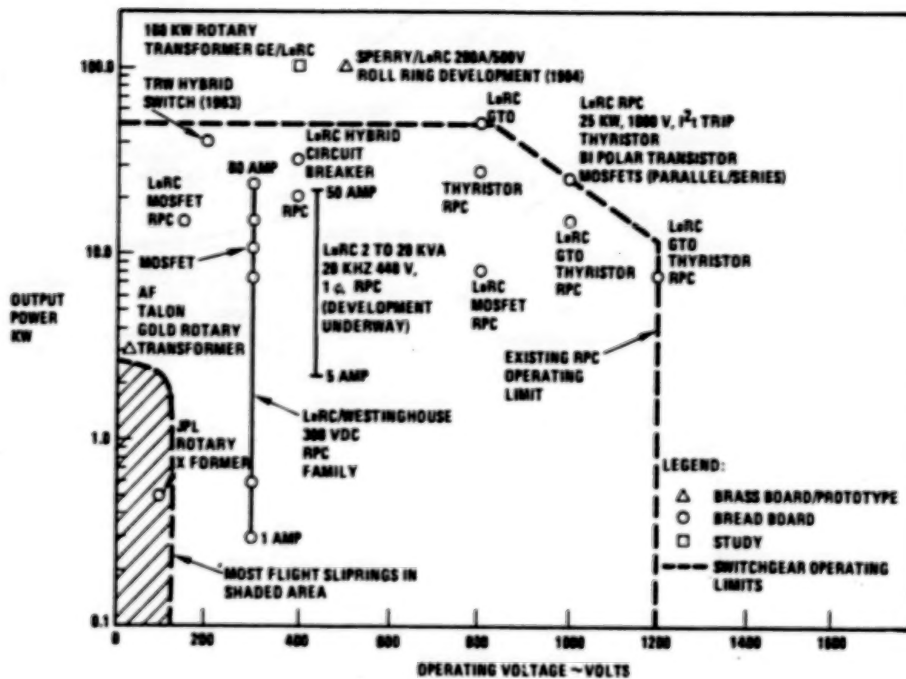


Figure 7. - High voltage distribution components.

ANALYTICAL MODELING OF SPACECRAFT POWER SYSTEMS

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NASA Goddard Space Flight Center

During the 25 years of space flight with unmanned earth orbiting satellites, there has been an evolution of power systems in three general areas. The size of power system in terms of power demand at the bus has increased from a few watts in the early 1960's to a few hundred watts during the 1970's. Today, the bus power requirements are typically in the .5 to 1kw range with some mission requirements exceeding the 1kw size.

The second evolution in spacecraft power has been the gradual increase in complexity, brought about by demand for more power and higher "fidelity" power. The most significant aspect of this is the high bandwidth systems (50Kz) that are designed to meet a wide range of payload requirements. Switching regulations of 20Kz are very commonplace in today's power conditioning equipment. Increase in bus power requirement has driven the size of solar arrays such that units of measure are in tens of meters square, making "all-up" end-to-end system tests impractical if not totally impossible. Consequently, as power systems have grown in size and fidelity, which inherently need higher levels of verification, the capability to perform a total system verification has become increasingly more difficult and costly. The extreme of this scenario is one where major power system components will have to be assembled and tested on the ground but due to physical size, the power system must be integrated on-orbit. The third area where evolution has had a significant impact on power system design is on the user side of the power bus. Today, there are individual science and application payloads that require more power than a total spacecraft provided during the 1960's. For instance, the Thematic Mapper (TM) on Landsat-D and D' requires 385 watts peak power while in the picture mode. Along with the higher power for instruments, has been the demand for very high fidelity power. This is due to a significant increase in sensitivity and/or resolutions of state-of-the-art detectors/sensors employed in these modern instruments.

Two recent examples of high fidelity power requirements include a 1mv regulation on a +15 volt power line and a noise requirement of less than $20\text{Nv}\sqrt{\text{Hz}}$ at 1Hz on a 5 volt power line. Although power grounding and user isolation are not subjects you hear or read about in symposiums, workshops, and conferences on power systems, these are major design considerations when high fidelity power is required by a user.

BACKGROUND

The need for and the use of power system models are not recent discoveries. Analytical models of an elementary nature have been developed for a number of spacecraft power system designs. In most cases, these models were derived out of the need to verify a particular performance observed on a

power system. Usually, the performance of the power system had already been established and a hybrid model (one using measured parameters combined with theoretical calculation) was used to determine margin and develop increased confidence in operating beyond the measured performance range.

The author's first experience with power system instability on orbit was on the Tiros-N spacecraft launched in October 1978. The oscillation occurred each orbit shortly after the batteries went into a current taper (voltage limit) mode and remained until spacecraft eclipse. The Tiros spacecraft and power system had been subjected to an extensive ground test program very similar to what other Goddard spacecraft and subsystems experience. In fact, the Tiros had a very strong design heritage in the Air Force Block 5-D program which had not experienced an oscillation problem. An investigation revealed an evolution of several design changes, any one of which was not considered to have a major impact on the systems phase and gain margins. Taken collectively, these changes resulted in a system that was marginally stable (or unstable) with the flight array connected and producing full power. Subsequent all-up system test of the power system on the follow-on satellite (NOAA-A) did reveal the oscillation when tested with an illuminated array, but only after all ground test instrumentation was disconnected from the power system.

The experience with the Tiros-N power system along with other similar problems observed on power systems during ground testing made it obvious that testing alone was not sufficient to assure the in-orbit integrity of a power subsystem.

In 1979, at the OAST Flight Technology Improvement Workshop (reference 1), the need for a high fidelity analytical model for power systems was identified as having highest priority. In 1980, NASA Headquarters funded a technology program at Goddard to develop comprehensive models of spacecraft power systems. The MSFC workshop on Space Power Systems Automation Technology (reference 2) held in October 1981, further highlighted the importance of accurate models to automation of power subsystems. The Goddard funded effort resulted in a TRW study (CR-166820) (reference 3) that reviewed the adequacy of existing models and recommended specific approaches to achieve the goals of the modeling effort. In 1983, a contract (NAS5-27543) was initiated with Lockheed to develop a small signal ac model of spacecraft power system(s). In parallel with the Lockheed development, an in-house effort was initiated to develop a dc energy balance model. The major part of the in-house effort is being devoted to the battery model.

GENERIC MODEL OF POWER SYSTEMS

The TRW study concluded that four models would satisfy the present and future needs for spacecraft power systems. These models are: system sizing and synthesis, dc small signal ac, and large signal transient. The sizing and synthesis energy balance model is primarily for performing system trade studies in terms of mass volume, area, and cost. Most cost models presently in use by industry are considered proprietary since the output may effect the competitive status. The sizing and synthesis models

range from "rule-of-the-thumb" approach to very detailed parameters for optimizing a system to meet a given requirement. The energy balance model's primary purpose is for performing energy calculations on an orbital basis. As an outgrowth of having an accurate energy balance model, the thermal dissipations associated with the various power system components is a by-product. By far, the greatest deficiency in existing energy balance models for low-earth orbit (LEO) is the lack of a good and reliable Ni-Cd battery model. Almost without exception, previous modeling efforts attempted to use battery state-of-charge for energy balance. Such models have proven to be very inadequate, especially in LEO where battery charging is usually accomplished with moderate to high initial rates followed by a constant voltage/current taper. One approach to battery energy balance modeling is to use "recharge ratio" which is how we actually monitor battery operations in-flight. Until a successful battery modeling is available, simulation of power systems using voltage-limit charging will continue to be done with a high degree of uncertainty in the results.

Small signal ac model is for determining system stability, by providing both system gain and phase margin. The system model is a composite of system component phase and gain margins. The past practice has been to perform fairly detailed phase and gain measurements on most of the electronic elements of a power system with little regard to the solar array, battery, and distribution (harness, connectors, relays, slip ring, etc.) elements of the system. However, as power systems have increased in size and bandwidth, the impedance characteristic of these elements now play a significant part in overall system stability. It should be noted as a matter of definition that small signal ac analysis is frequency domain where linear methods of analysis apply.

The large signal transient model deals primarily in the time domain from milliseconds to kiloseconds. Analysis in this region deals in the nonlinear region of system and component operations that occurs during mode changes, load switching, and system faults. Unlike the small signal modeling, transient modeling is virgin territory since there is little theoretical or analytical background in this area. This model is anticipated to be the most difficult to complete.

Figure 1 depicts the relationship between NASA program development phases and the four power system models. Also indicated, are the various program activities during each phase where the models will be used. The importance of the d.c. energy balance model is evident in that it is needed during all program phases. Figure 2 illustrates the overall Goddard approach to developing the power system models. The models described previously will be developed for each component. There will be an accompanying data base with each component model. The data base will be flexible in that the user may augment it with his own data. The system model provides for tying the component models together into a configuration designated by the user. The overall program will be run by an executive program identical to or similar to the Integrated Analysis Capability (IAC) (reference 4) already in existence.

The IAC Executive is illustrated in figure 3. The IAC already has the

capability to operate NASTRAN and other large programs containing large data bases. It is envisioned that the power management and distribution (PMAD) program will become another element that uses the IAC as an Executive driver.

In summary, the development of comprehensive models of space power systems is clearly mandated if we are to meet the current and future design challenges brought about by the demand for higher power, high speed, automated systems. The major attributes of these models are commonality and compatibility, and modularity at component level with sufficient confidence to scale to any power level. Equally important to these models is verifiability, user friendly, and portability. The latter two attributes are mandatory if the models are to gain wide acceptance by industry and government power system designers.

REFERENCES

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2. NASA Conference Publication 2213, Space Power Subsystem Automation Technology, Proceedings of a Workshop held at the Marshall Space Flight Center, Alabama, October 28-29, 1981.
3. NASA Contractor Report (CR-166820), Final Report, Analytical Model of Spacecraft Power Systems, April 1982, John Cassinelli.
4. Development and Use of an Integrated Analysis Capability, R. G. Vos, et al. Boeing Aerospace Company, and J.P. Young, H. P. Frisch, NASA/Goddard Space Flight Center, AIAA 1983.

NASA FLIGHT PROGRAM DEVELOPMENT & POWER SYSTEM MODELS

PHASES/FUNCTIONS

MISSION ANALYSIS (PHASE A)
TRADE STUDIES
COST, MASS, VOLUME

DEFINITION (PHASE B)
SYSTEM DESIGN & SIZING
SYSTEM OPTIMIZATION
SYSTEM MARGINS
SYSTEM INTERACTIONS

EXECUTION (PHASE C)
DETAIL SYSTEM DESIGN
HARDWARE DEVELOPMENT
COMPONENT VERIFICATION
SYSTEM INTEGRATION
PERFORMANCE VERIFICATION
SYSTEM INTERACTION

OPERATIONS (PHASE D)
ORBITAL PREDICTION
ENERGY/POWER MANAGEMENT
PAYLOAD INTERACTIONS
ANOMALY/FAILURE ANALYSIS

SYSTEM MODULES

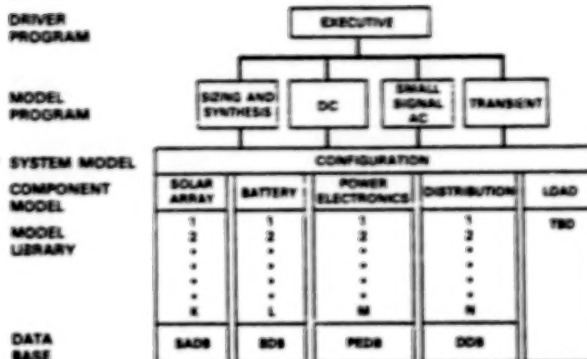
SYSTEM SIZING AND SYNTHESIS
D.C. ENERGY BALANCE

D.C. ENERGY BALANCE
SYSTEM SIZING AND SYNTHESIS

SMALL SIGNAL A.C.
LARGE SIGNAL TRANSCIENT
D.C. ENERGY BALANCE

D.C. ENERGY BALANCE
LARGE SIGNAL TRANSCIENT
SMALL SIGNAL A.C.

APPROACH TO POWER SYSTEM MODELING



NOTE: MODEL VERIFICATION BY USING ON ONGOING FLIGHT PROGRAMS

Figure 1.

Figure 2.

EXECUTIVE PROGRAM

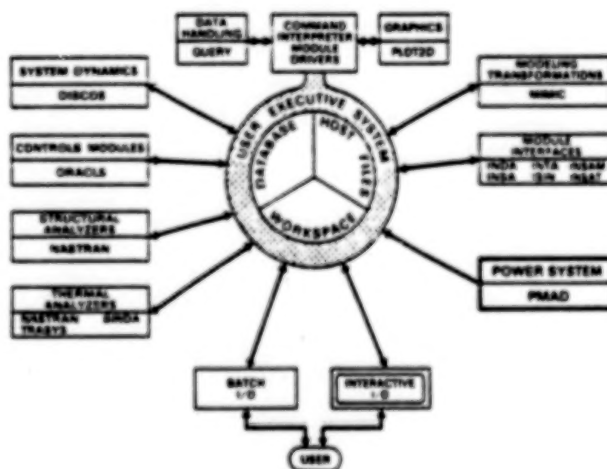


Figure 3.

EFFECTS OF THE SOLAR AND EXTRATERRESTRIAL ENVIRONMENTS
ON SPACE POWER SYSTEMS*

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California Institute of Technology

Environments surrounding the major extraterrestrial bodies in the Solar System and their interactions with spacecraft power systems are summarized. The environments associated with neutrals/dust, low energy plasma, and where applicable, magnetospheres are discussed for a wide variety of cases. The impact of these environments on power systems--in particular, radiation effects, spacecraft charging, plasma interactions, surface sputtering/erosion, and induced currents--are presented. As power systems must be designed to survive in these hostile environments, it is important that they be taken into account in planning future power systems.

1. INTRODUCTION

To date the vast majority of all space missions has been flown in that near-Earth region of space called the magnetosphere. Increasingly, however, with missions to the Moon and outer planets, extraterrestrial environments are posing interesting and often severe constraints on space power systems. The objective of this paper will be to briefly review those aspects of extraterrestrial environments that are of concern to power system engineers. Of particular concern will be the environments associated with neutrals/dust surrounding the body, the low energy plasma, and, where applicable, the magnetosphere of the extra-terrestrial body. These environments give rise to five interactions of direct concern to power systems. These are: radiation effects, spacecraft charging, plasma interactions (i.e., power loss, enhanced arcing, etc.), surface sputtering/erosion, and induced currents. While no attempt will be made to analyze these effects in detail, where possible they will be described for each of the bodies studied. Finally, it should be kept in mind that much of the environmental data presented are of a preliminary nature--in several cases no in-situ data exist and theoretical extrapolations must be used. Even so, the data presented do give valuable insights into the problems power system engineers are likely to encounter in the design of future systems required to survive in these exotic environments.

*This paper presents the results of one phase of research carried out at the Jet Propulsion Laboratory, under contract NAS-7-918, sponsored by the National Aeronautics and Space Administration

2. THE INTERACTIONS

This section summarizes the major spacecraft interactions of importance to the power system engineer. The objective is to provide an overview of the more critical interactions and to provide simple quantitative tools for estimating their effects. While there is no attempt to make this a detailed review, it is fairly comprehensive. As will become apparent, the interactions are grouped in terms of the neutral atmosphere, ionosphere, and magnetosphere. This pattern will be followed in the subsequent review of the planetary environments.

By far the major environmental factor at low altitudes is the ambient neutral atmosphere. Whether it be through drag or the recently discovered interactions with atomic oxygen, the effect of the neutral atmosphere (predominately the neutral atomic oxygen) on spacecraft dynamics and surfaces greatly exceeds any of the other effects that will be considered in this report. The source of the neutral atmosphere interactions at low altitudes results of course from the direct impact of neutral particles on spacecraft surfaces. This causes drag and surface damage/abrasion. The standard expression for the drag force is:

$$\begin{aligned} F(\text{drag}) &= 1/2 \rho V^2 CD A = \\ &= 300 - 5000 \text{ dynes for the shuttle} \end{aligned} \quad (1)$$

where:

- ρ = density (typically 10^{-15} g/cm³ near the Earth)
- CD = drag coefficient = 2.2 - 4.0
- A = cross-sectional area of spacecraft
= 50-400 m² for shuttle
- V = spacecraft velocity
= 7.6 km/s (for LEO)

Given the importance of ionospheres to radio and radar propagation, it is not surprising to find that models are available for most of the planets. However, most of these models only predict electron densities--the most readily measureable quantity and the most important to radio propagation--whereas the ion densities are often estimated from theoretical models. Ionospheres are also, because of the high plasma density associated with them, a primary source of interactions with power systems. In particular, the high plasma densities can cause spacecraft charging (generally weak, however), increased arcing at high applied voltages, and power loss. On the basis of simple models for ion collection (described in reference 1), potentials for various ionospheric conditions throughout the solar system have been estimated and are presented in table 1. The spacecraft-to-space potential varies from a few tenths of a volt in the ionospheres to a few hundreds of volts in the solar wind (the Sun's ionosphere)--in rough agreement with observations (ref 1).

While charging is relatively weak, the high densities will, however, encourage plasma interactions with exposed high potential surfaces. Although surfaces may be insulated, even small pinholes or scratches that penetrate the insulation can, in the case of positively biased surfaces, enhance electron collection so that the insulation is useless. This leads to charge collection and, for typical ionospheric densities, power loss. Estimates of this power

loss range as high as several 10's of percent for solar arrays with voltages in excess of a few 100's of volts immersed in a low energy ionospheric plasma like that of the Earth's.

Although the details of magnetospheres (i.e. the locations of radiation belts, etc.) are not well defined, the geomagnetic field is generally known for most of the planets and moons. Besides magnetic torques (which are very system dependent), such magnetic fields can induce an electric field in a large body by the $v \times B$ effect:

$$E = 0.1 (v \times B) \text{ V/m } (= .3 \text{ V/m for LEO}) \quad (2)$$

where:

v = spacecraft velocity (= 7.6 km/s LEO)

B = magnetic field (= .3 G for the Earth's equator)

In a later section, the maximum induced electric field (that observed in low altitude orbit over the equator) will be estimated. As a practical example, for a spacecraft like the shuttle (roughly 15 m x 24 m x 33m), potentials of 10 V could be induced by this effect. As power systems grow to the scale of kilometers or larger, the induced fields will grow accordingly. These fields are comparable to the fields necessary to deflect charged particles in the ionospheric environments since the particles have ambient energies of ~ 0.1 eV. Thus, these fields must be taken into account in the study of ionospheric fluxes. It should also be noted that for planets with strong magnetospheres like Jupiter and the Earth, ambient electric fields may approach or exceed these induced fields on occasion.

Spacecraft charging is the result of nature's attempt to bring about current balance on spacecraft surfaces. The major source of current to a spacecraft is the ambient electron population. As the ion flux is usually an order of magnitude smaller, other currents generally provide the balancing current. Typically, in the inner solar system, the photoelectron flux equals or exceeds the electron flux so that, in sunlight, the spacecraft floats a few volts positive (roughly the average energy of the photoelectrons). In eclipse, the secondary electrons given off by electron impact normally balance the ambient electrons. Potentials between the spacecraft and the ambient environment can, however, reach as high as several KV (the characteristic energy of the ambient electrons) during such periods. In principle, for a large structure, the shadowed portion could build up such large potentials relative to the sunlit side. It is these differential potentials between spacecraft surfaces rather than the spacecraft-to-space potential that cause arcing. The ranges for such potential differences are estimated for various magnetospheres in Table 1. In contrast to the potentials observed in low altitude ionospheres, large potentials can be attained in the outer magnetospheres of several of the planets.

The final class of effects that need to be considered for the outer planets are those due to radiation. Besides the familiar long term radiation damage to IC's and solar cells, the less well known effects of internal charge deposition and single event upsets must also be considered. Although not treated extensively in this short presentation, these latter effects are currently causing serious problems for system designers. Charge deposition resulting from the penetration and deposition of electrons on and in interior

spacecraft surfaces may have caused 42 anomalies on Voyager 1 at Jupiter. Single event upsets, the results of cosmic rays penetrating the sensitive regions of IC's and flipping their logic states, are now blamed for several serious logic resets in spacecraft command circuits. As these effects are intimately linked to the control circuitry for power systems they, too, must be considered in power system design. Here we will touch on these effects briefly when discussing the jovian environment.

3. SOLAR ENVIRONMENT

The dominant environment in the Solar System is that of the solar atmosphere or heliosphere--the Solar Wind. This is the low density plasma (predominately hydrogen ions with some helium) that is continuously emitted from the solar corona at supersonic speeds. The plasma is characterized by a residual magnetic field (typically a few 10^3 's of nT [$1\text{ nT} = 1$ nano Tesla or 1 gamma]) and variable velocity and density. The Solar Wind velocity vector is observed to be dominantly in the radial direction in the ecliptic plane with a magnitude of 200 to 500 km/sec. Since the Sun rotates with a period of 27 days, the Solar Wind takes on a spiral structure as illustrated in Fig. 1 with the spirals marked by regions of similar magnetic polarity. At present, based on in-situ measurements from the Pioneer spacecraft, we know that this environment extends out to and beyond the orbits of Pluto and Neptune, where at some point it terminates in the interstellar medium (Fig. 1b). Typical plasma values for the Solar Wind are summarized in Table 1 for distances corresponding to near Mercury, the Earth, and Jupiter. As will be discussed later, the Solar Wind represented by these values can be a significant source of spacecraft charging.

Aside from the plasma, the Solar Wind can exhibit large variations associated with solar flares and shock waves. While the dominant cold plasma may increase three- or fourfold, the high energy fluxes ($E > 100$ keV) associated with solar flares can increase many orders of magnitude in a relatively short time. Example of several severe flares are presented in Fig. 2. Although such severe flares are relatively infrequent, by far the greatest threat to any space system are the radiation effects associated with these flare particles and they must be seriously considered in the design of power systems for long duration space missions (refs. 2 and 3).

4. MERCURY

Mercury, long thought to be a relatively dead, lifeless world much like the Moon, has presented several surprises. While it is true that in many respects Mercury does closely resemble the Moon, in one very important aspect it differs significantly--it has a magnetic field. As a result, the environment near Mercury is dominated as in the Earth's case by its magnetic field. This field structure is illustrated in Fig. 3. The effects of this field will be compared with the other planets in a later section. Unfortunately, due to the sparsity of data, little can be said of the magnetospheric/ionospheric environment enclosed by this field except that it is shielded from the Solar Wind. Table 2 lists some upper limits on the ionosphere and atmosphere of Mercury (refs. 4 and 5).

5. VENUS

Of great interest as "twin" to the Earth, Venus turns out to be relatively dull as magnetospheres go. Apparently because of its extremely slow rotation rate (roughly equal to its year), it possesses no significant intrinsic magnetic field. Its interaction with the Solar Wind is that of a blunt obstacle. The Solar Wind interacts directly with the ionosphere of Venus as illustrated in Fig. 4a. This leads to compression of the ionosphere on the dayside of Venus in comparison to the nightside (illustrated in Fig. 4b). As the neutral atmosphere of Venus is dominated by CO_2 , it is not surprising that oxygen and carbon ions dominate the ionosphere and atmosphere at high altitudes (Figs. 4b, c). Given the problems with atomic oxygen erosion and glow at the Earth for the shuttle, it is obvious that power systems (particularly solar arrays) will have to be protected from these problems at Venus also (refs. 6-8).

6. MOON

The largest data base we have on an extraterrestrial body is that concerning the Moon. Various lunar orbiters have allowed a detailed mapping of the lunar magnetic field and surface instruments have allowed long term, in-situ observations of the lunar ionosphere/atmosphere. The Moon, for all practical purposes, however, is devoid of a magnetic field. Rather, the lunar magnetic field is, as illustrated in Figs. 5a, b, characterized by local magnetic anomalies of a few 10's to 100's of nT and the effects of magnetic induction generated by the Solar Wind, which penetrates almost unhampered to lunar surface (note: the lunar environment is that of the Earth's at 60 R_E when the Moon is in the magnetosphere). The Moon, as does Venus, creates a void in the antisolar direction (Fig. 5a) into which the Solar Wind eventually expands due to diffusion. For power systems, then, the environment at the lunar surface is that of the Solar Wind, the Earth's magnetosphere, or a region devoid of plasma depending on the position of the Moon in its orbit. The only other environmental concern of note is the apparent existence of an electrostatically levitated dust layer near the solar terminator (refs. 9-11).

7. MARS

Thanks to the numerous probes to Mars and the Viking landers, there is a fairly detailed understanding of the martian environment. Unfortunately, there is still considerable controversy over the existence of a martian magnetic field. If it exists, it is very small (see Table 3) so that the magnetosphere of Mars resembles that of Venus and the Moon. This magnetosphere is modelled in Fig. 6a. Models of the martian ionosphere and atmosphere are presented in Figs. 6b, c. As for Venus and the Earth a principle concern in the martian environment will probably be oxygen contamination since the density between 200 to 400 km resembles that of the Earth. This may well turn out to place a limit on the orbital altitude of missions with systems sensitive to this problem. Also, as in the case of both the Moon and Venus, solar flare particles will be little deviated by the martian magnetosphere so that they may pose a threat to orbiting vehicles (refs. 12-14).

8. JUPITER

After the Sun, the Solar System is dominated by Jupiter. Its magnetic field is the largest in the solar system and dominates the space around it out to apparently the saturnian orbit. High energy particles from Jupiter have been observed at the Earth. It is no wonder then that the interactions with Jupiter's environment are the most significant in the Solar System. Like the Earth, however, the jovian magnetosphere is extremely complex and marked by regions of very pronounced variations. Here only a few of them will be described and then only in cursory detail.

The magnetosphere of Jupiter is dominated by three factors: the intense jovian magnetic field (100,000 times that of the Earth) and its tilt relative to the jovian rotation axis (11 degrees), the rapid rotation of Jupiter (10 hours), and the jovian moon Io. The jovian magnetic field is so strong that it controls space out to $80 R_J$ ($1 R_J = 1$ jovian radius) in the sunward direction. Io generates a vast torus of neutral gas (visible from the Earth) around Jupiter at $5 R_J$. The rapid rotation rate of Jupiter's magnetic field forces the cold plasma associated with this torus to expand by centrifugal force into a giant disc. The tilt and the rotation rate make this plasma disc wave up and down so that at a given location plasma parameters vary radically over a 10 hour period. Superimposed over the plasma disc is an intense radiation belt that resembles nothing less than a nuclear burst environment. Each of these features is illustrated in Fig. 7a.

Based on the preceding, Jupiter's environment can be divided into roughly three populations: the cold plasma associated with the Io torus and the plasma disc ($0 < E < 500$ eV), the intermediate plasma ($500 \text{ eV} < E < 100$ keV), and the radiation environment ($E > 100$ keV). The cold plasma environment is characterized by high densities (Fig. 7b) and low temperatures. The plasma near the torus and the disc consists of hydrogen, oxygen (singly and doubly ionized), sulfur (singly, doubly, and triply ionized), and sodium (singly ionized) ions. The ions are forced to corotate with Jupiter out to well past $20 R_J$. Moderate energy particles (electrons and protons) are not well modelled. Estimates of their densities are presented in Fig. 7c. Typical temperatures are 30 keV for the protons and 1 keV for the electrons. Contours for the electron and proton fluxes above 1 MeV are shown in Figs. 7d, e. These fluxes are the most intense in the Solar System at energies above 1-2 MeV. For completeness, representative models of the ionosphere and atmosphere of Jupiter are presented in Fig. 7a.

Clearly there are many problems for power systems at Jupiter. The high density of the Io torus possesses plasma interaction problems for exposed biased surfaces. Likewise, spacecraft charging problems are also a possibility. Contours of equal spacecraft-to-space potentials are presented in Fig. 7f (see reference 15 for an explanation of the model used). Although these values agree with actual observations, as we do not yet know the details of the variability of the jovian environment it is not possible to rule out even larger values. Similarly, the harsh radiation levels imply severe radiation damage and the possibility of internal spacecraft charging. The presence of trapped high energy protons and heavy ions raises serious concerns about soft logic upsets in control computers. For these reasons, great care and expense have been taken in designing the forthcoming Galileo spacecraft to survive in this environment, (refs. 15-18).

8. IO

Because of its singular scientific value (it has the only known active volcanoes aside from the Earth's in the solar system and generates a huge plasma torus) Io has been singled out as a flyby target for future missions. Values for its atmosphere, the torus, and its ionosphere are presented in Table 4. Further, the Voyager spacecraft are believed to have detected a magnetosphere and, indeed, calculations of the shape of such a minimagnetosphere have been attempted. Also, because the magnetic field of Jupiter is so strong at Io, it is believed that a tremendous electric field is set up between the planet and its moon due to $v \times B$ forces. These effects, it has been conjectured, may make the environment around Io dangerous to spacecraft. At this time, however, the calculations are too rudimentary to be of value (refs. 19 and 20).

9. SATURN

Saturn is marked by a magnificent set of rings that are its most obvious feature and set it aside from all the other planets (Jupiter and Uranus both apparently have small ring systems). For the purposes of this interaction study it is interesting to note that it has been conjectured that the ring particles may be charged (this is evidenced by the variations in the ring "spokes" observed by Voyager). Aside from the rings, however, Saturn resembles Jupiter. Like Jupiter it has an extensive magnetosphere and radiation belts. These are represented in Fig. 9a. Unlike Jupiter, however, Saturn's magnetic field axis is apparently aligned with the spin axis so that the plasma ring around Saturn is relatively stable compared to that of Jupiter. Compared with the intense environments around Jupiter, Saturn's appears relatively benign. Even so, the cautions concerning Jupiter also hold at Saturn, and vehicles should be designed for a relatively harsh radiation environment. The ionospheric and atmospheric environments are illustrated in Figs. 9b, c.

10. TITAN

Titan is the only planetary satellite in the Solar System known to have a substantial atmosphere. Although the primary constituent is N_2 , the surface is totally obscured at optical wavelengths by layers of haze and methane clouds. Understanding of the composition and structure of the titanian atmosphere was considerably advanced by Voyager IR spectrometer data. Fig. 10 shows the vertical temperature profile as derived from Voyager IR and radio occultation measurement. Table 5 shows the trace composition of the atmosphere as derived from Voyager IR data. Because of the presence of organic molecules, the titanian atmosphere is a subject of intense scientific interest (ref. 24).

10. URANUS

In 1986, Voyager 2 will pass near Uranus, providing us with our first close views of this distant gas giant. In many ways, it is anticipated that Uranus will be quite surprising. Like Jupiter, Uranus apparently has several weak rings. Unlike Jupiter and Saturn, the rotational axis of Uranus and that of its magnetic field are both inclined about 90 degrees to the ecliptic plane! In 1985, the poles of Uranus will be lined up along the Uranus-Sun line. This makes for a very unusual magnetosphere as illustrated in Fig. 11a. Because of the remoteness of Uranus, little is now known about the environment

around Uranus. Some representative models of the ionosphere and atmosphere are illustrated in Figs. 11b, c. The green color of Uranus is due to strong absorption by methane in the atmosphere. Given the paucity of data on Uranus, little can be predicted about interactions with power systems in its environment (refs, 25-27).

11. COMETS

Shortly, several spacecraft from many different nations will encounter comets for the first time. Estimates of the environments around comets indicate that, from an interactions standpoint, they will be the most interesting planetary bodies to have been studied to date. This is because of the structure of the comets. Based on current models (see Figs. 12a, b, c), comets are pictured as consisting of a snowball-like center a few kilometers in diameter. This nucleus consists of water-ice, other gaseous materials, rocks, and dirt. As the comet approaches the Sun, the ice and frozen gases boil off and the dirt or dust is blown away by the solar light pressure. The rocks remain in orbit and may eventually contribute to the meteor flux at the Earth. The gas and water vapor are ionized and, as they move away from the comet, eventually become controlled by the Solar wind. Near the comet nucleus, however, the gas and dust dominate and form a bright region called the coma. Thus, the comet can be envisioned as having a miniature magnetosphere. Trailing away from the comet in the anti-sunward direction is a trail of dust. Separate from this dust tail is the plasma tail controlled by the Solar Wind and subject to its vagaries.

For those missions intending to penetrate the coma or portions of the tails, very real interactions problems exist. First, because of the tremendous relative velocities (upwards of 100 km/sec) involved in comet intercepts, the dust particles and neutral gas particles pose serious threats to the physical integrity of the spacecraft. The larger dust fragments may be able to penetrate surfaces while the smaller dust grains and the neutral particles may sputter surfaces and cause charging due to impact ionization. In Fig. 12b, the potential contours around the Giotto spacecraft are presented. Although the potential levels pose little threat to the Giotto mission, they threaten to compromise any direct measurements of the plasma and ambient environment encountered by the vehicle in the vicinity of the comet. Likewise, any exposed solar arrays, if used on a comet mission, would be subject to erosion and possible plasma interactions as a result of the plasma created when the comet material interacts with the leading surfaces. Landing on a comet nucleus as has been proposed will place additional constraints on the power system selected (refs 28, 29, 30, 31).

12. CONCLUSIONS

The findings of this study are summarized in Tables 1 and 3 and in Figs. 13a, b. In Table 1 are listed several of the plasma environments described above. As outlined in ref. 1, the spacecraft-to-space potentials under a variety of assumptions have been calculated. It is clear that the major spacecraft charging threat is represented by the Earth, followed by Jupiter and the Solar Wind. Given the pervasiveness of the Solar Wind, this may be the greatest of the spacecraft charging threats (particularly when solar flare effects are considered).

In Fig. 13a, the ionospheres of several of the planets are compared. Plasma interactions are generally associated with high density as this can

serve to short out exposed potentials on solar arrays or other exposed high voltage surfaces. From that standpoint, Venus, Earth, and Jupiter all pose threats to high power systems in some of their magnetospheric/ionospheric regions.

Another way of comparing the interactions around the planets is to compare their magnetic fields and magnetospheres. This has been done in Table 3 and Fig. 13b. In Fig. 13b, the Solar Wind standoff points for the various magnetospheres have been equated and the planetary radii plotted to scale. This clearly demonstrates the significant differences between Mars and Venus and the planets with strong magnetic fields--Jupiter, Earth, and Saturn. As another demonstration of this, in Table 3 the $v \times B$ electric field for a body in orbit at the surface of a given object is presented (this is presumably the maximum induced field that a system would see). Again, the Earth, Jupiter, and Saturn have the only significant fields (the values for Uranus are questionable).

To summarize, when compared with the other environments in the Solar System, the three dominant ones are Jupiter, the Earth, and the Solar Wind. These environments all pose threats to future power systems through the effects of radiation, spacecraft charging, and plasma interactions. The presence of atmospheres at Venus, the Earth, and comets pose threats to future power systems through the effects of radiation, spacecraft charging, and plasma interactions. The presence of atmospheres at Venus, The Earth, and comets pose possible erosion/sputtering threats. Around Jupiter, the Earth, and Saturn $v \times B$ effects are possible. At the very least, power system engineers must be aware of these problems and plan for their systems to survive in these hostile environments--environments in some cases that resemble the aftermath of nuclear warfare.....

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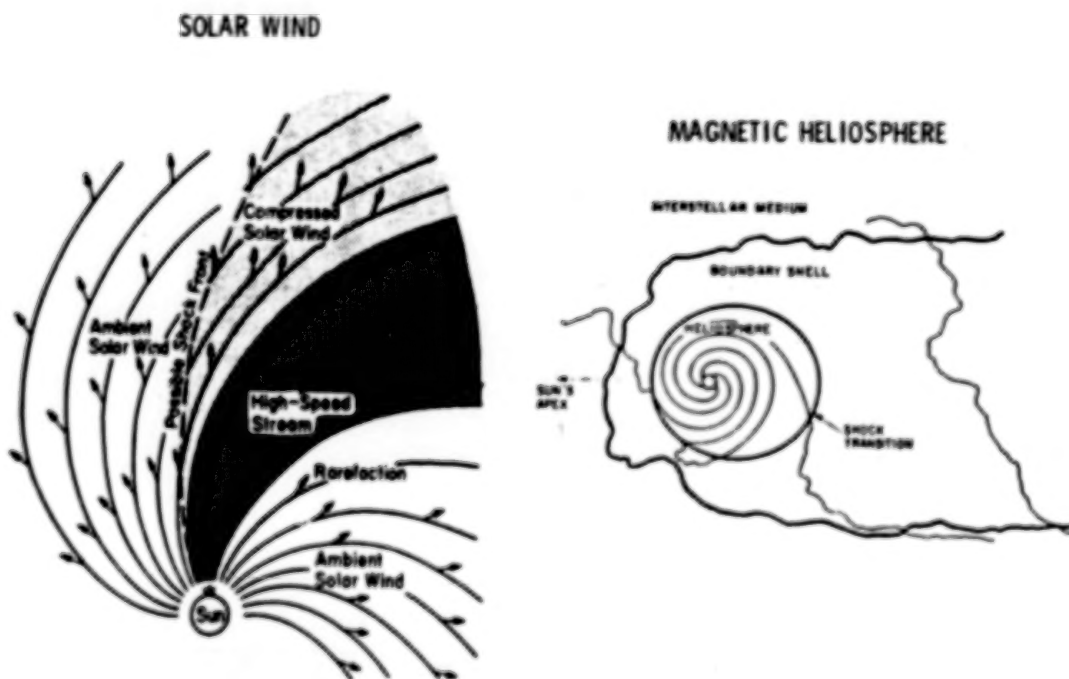


Figure 1. - Spiral structure of the Solar Wind (refs. 2, 3).

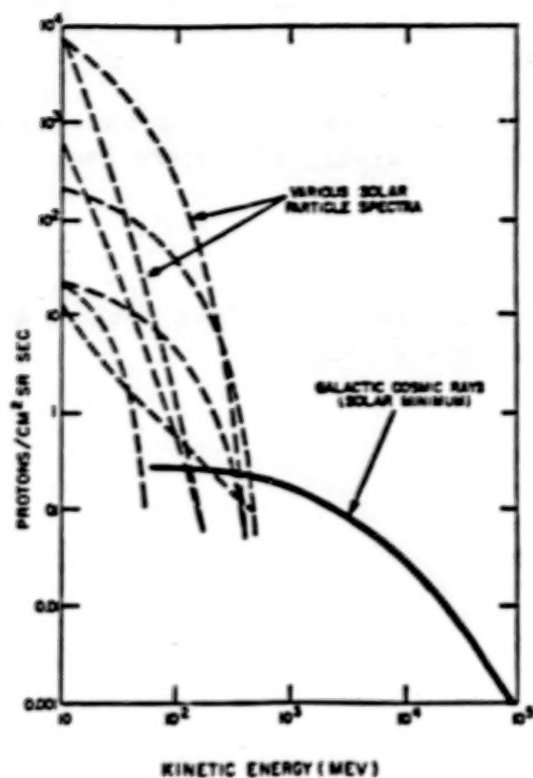


Figure 2. - Particle spectra for selected solar flares compared to galactic cosmic ray proton spectrum at solar minimum (ref. 33).

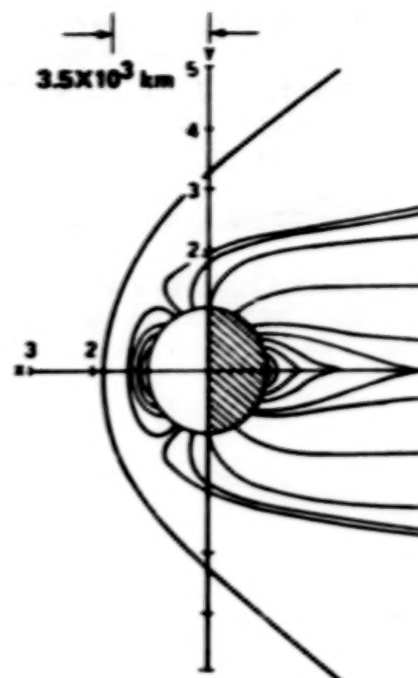
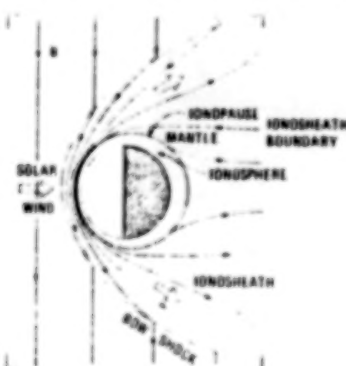


Figure 3. - Field structure of the magnetosphere of Mercury (ref. 4).

SOLAR WIND-IONOSPHERE INTERACTION



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IONOSPHERE

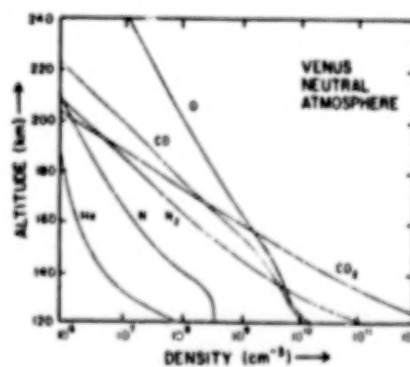
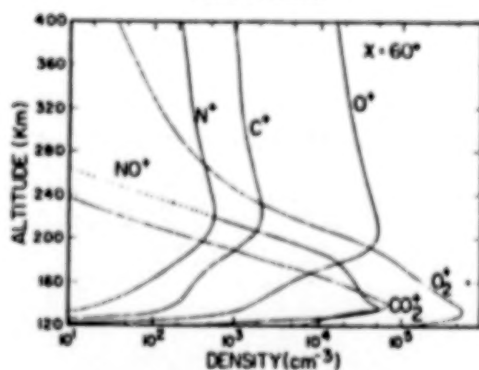
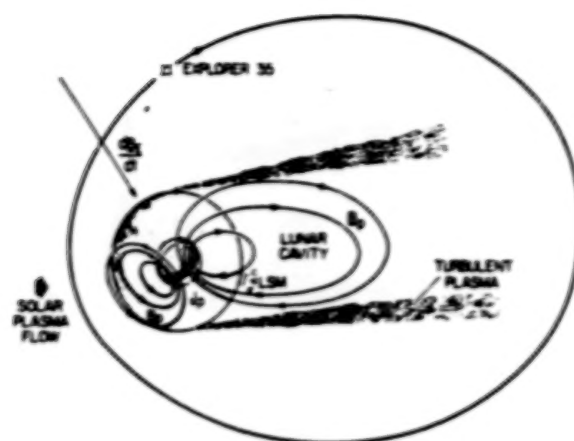


Figure 4. - Venus environmental models (refs. 6 to 8).



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Figure 5a. - Interaction of the Moon with the Solar Wind (ref. 9).

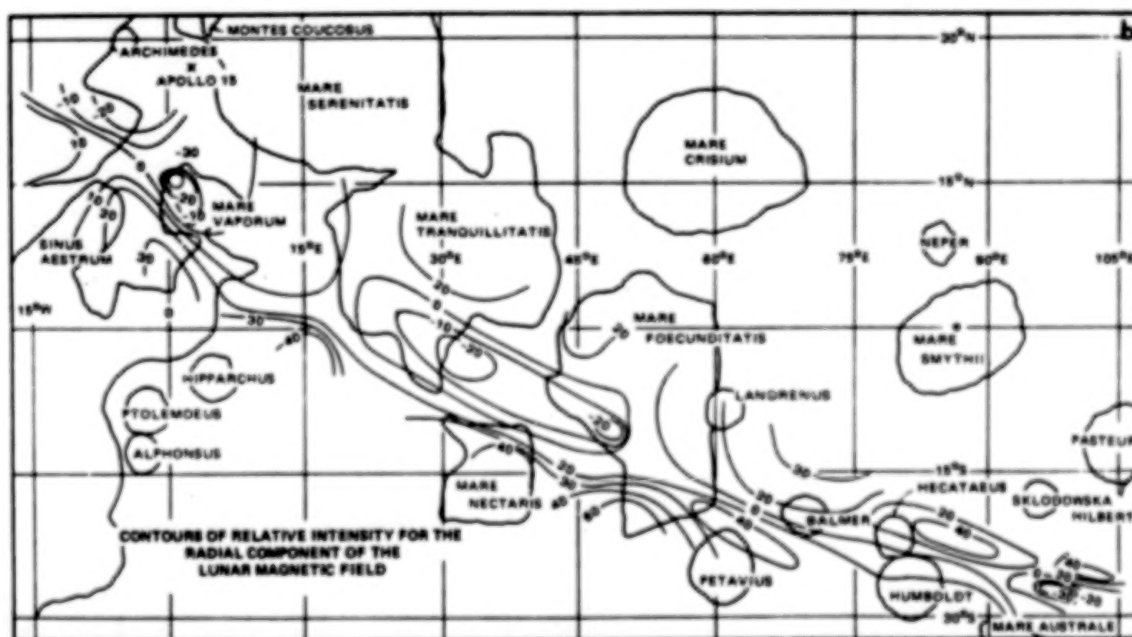
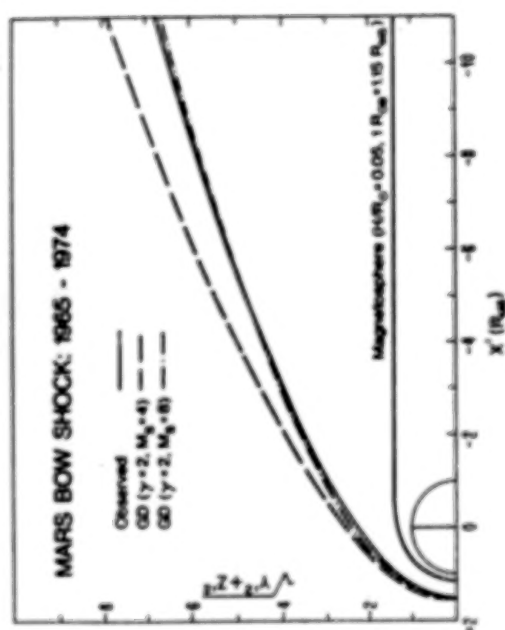
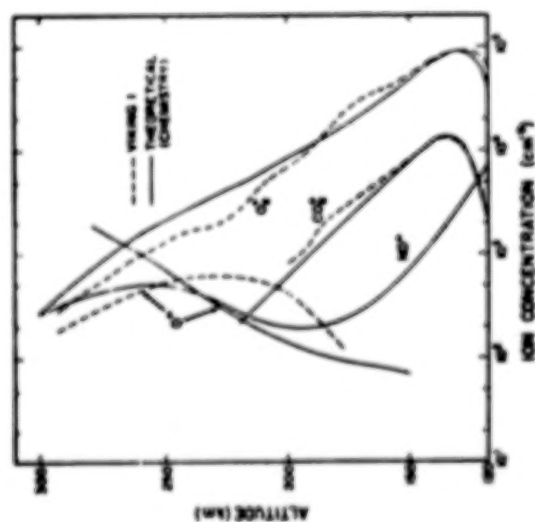


Figure 5b. - The Moon's local magnetic fields (ref. 10).

MAGNETOSPHERE



IONOSPHERE



NEUTRAL ATMOSPHERE

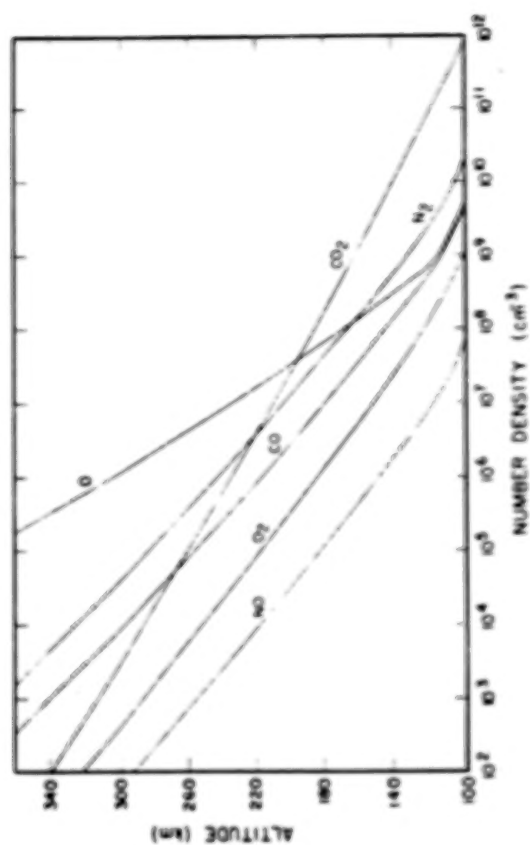
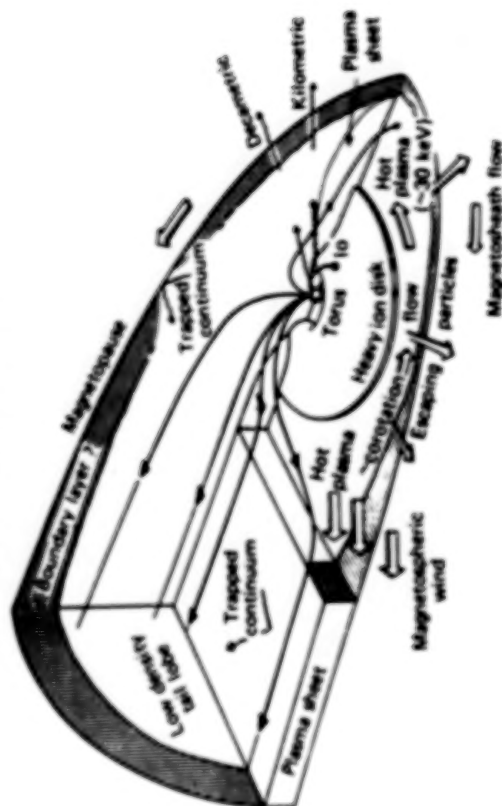


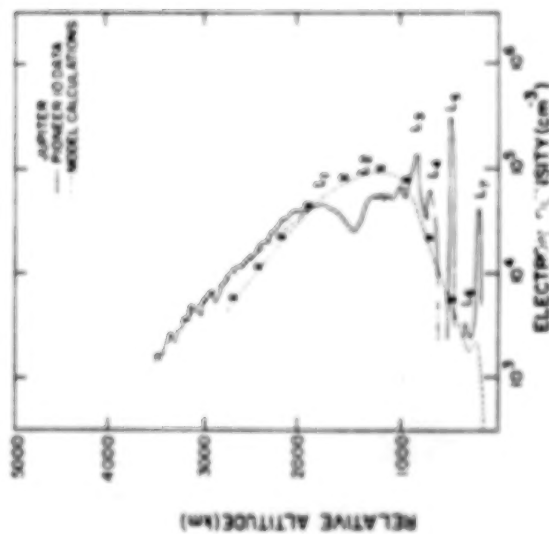
Figure 6. - Mars environmental models (refs. 12 to 14).

MAGNETOSPHERE



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IONOSPHERE



NEUTRAL ATMOSPHERE

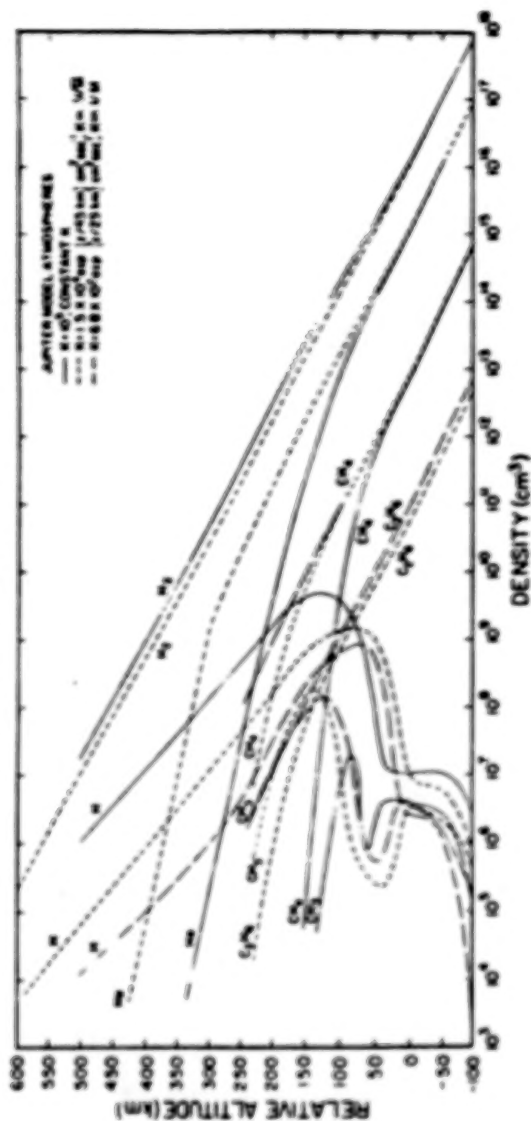
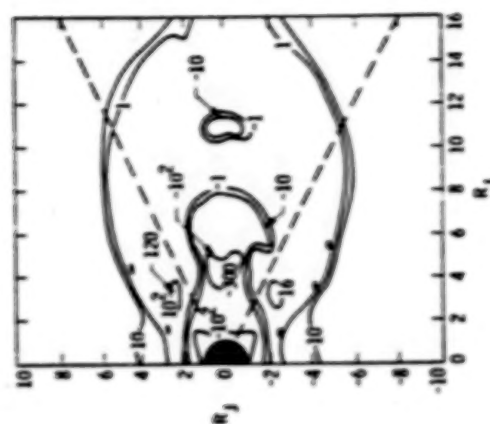
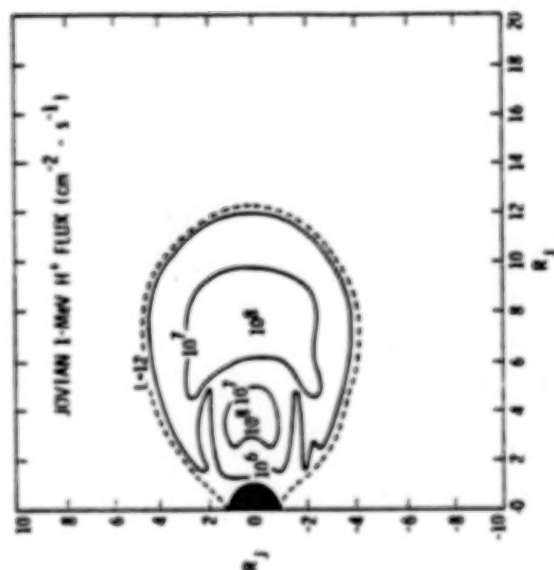
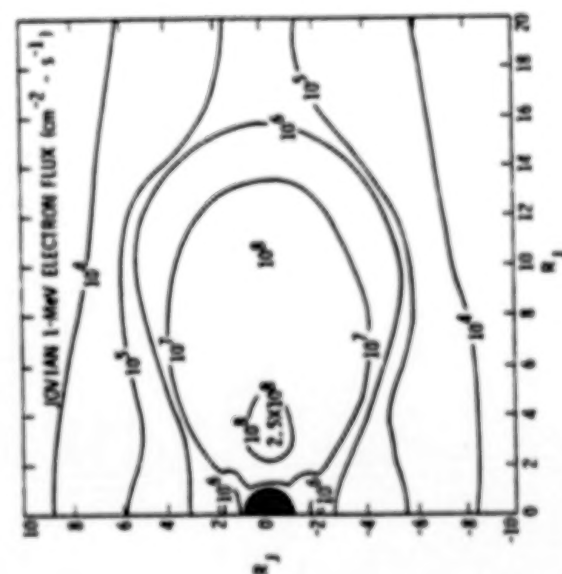
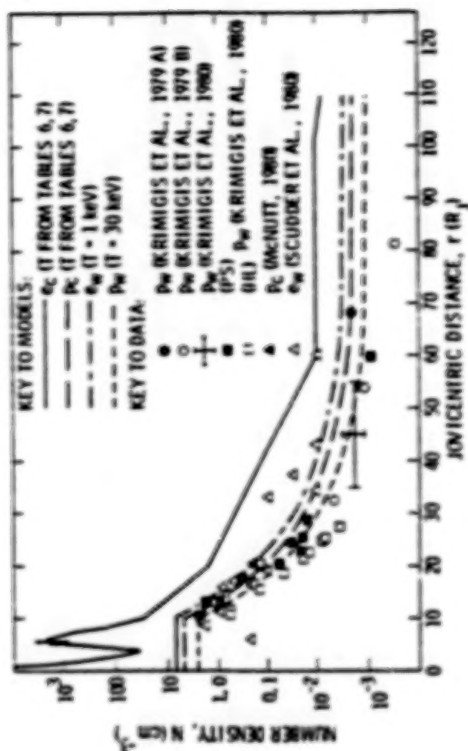
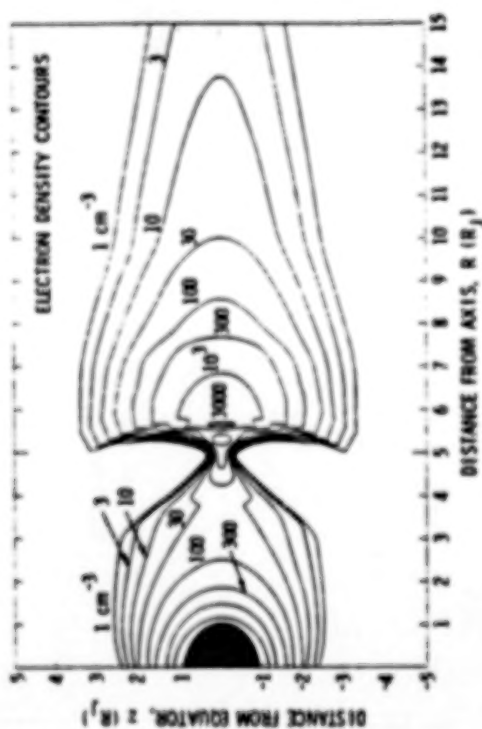


Figure 7a. - Jupiter environmental models (refs. 16 to 18).



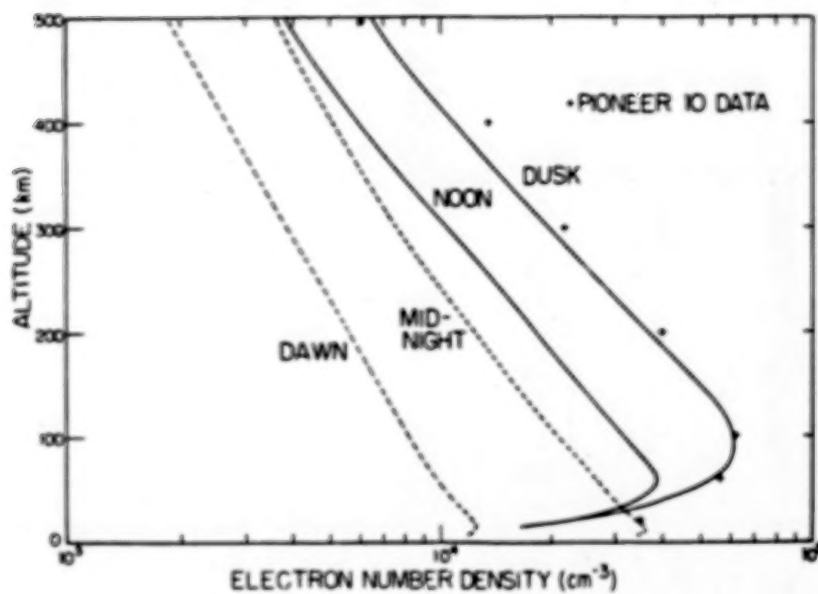
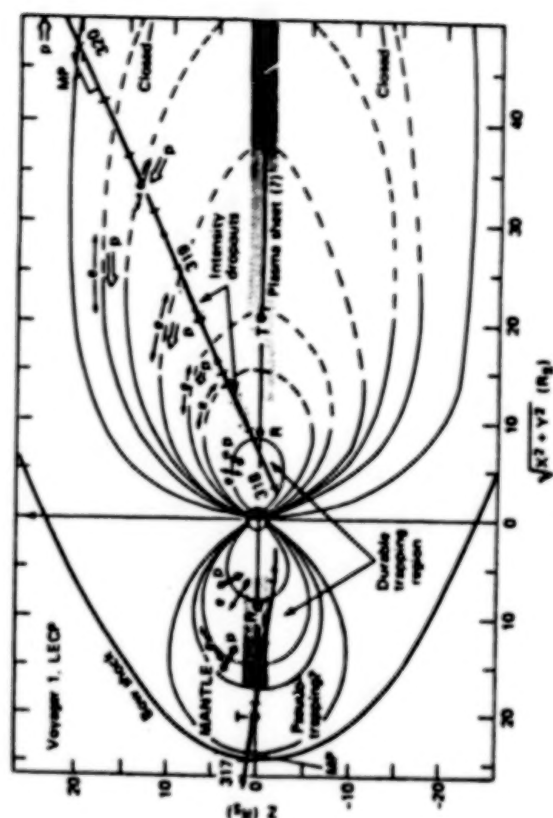


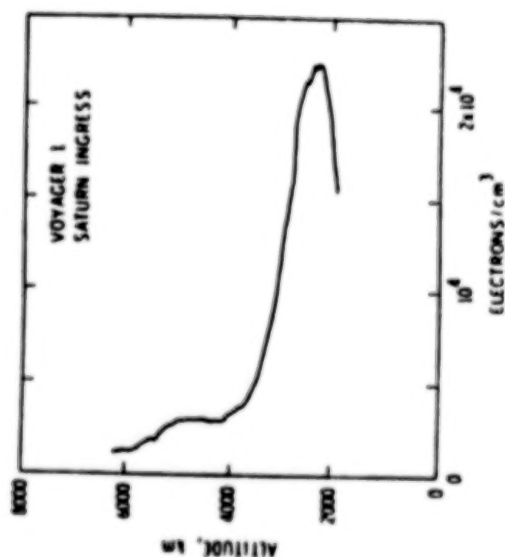
Figure 8. - The ionosphere of Io (ref. 20).

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MAGNETOSPHERE



IONOSPHERE



NEUTRAL ATMOSPHERE

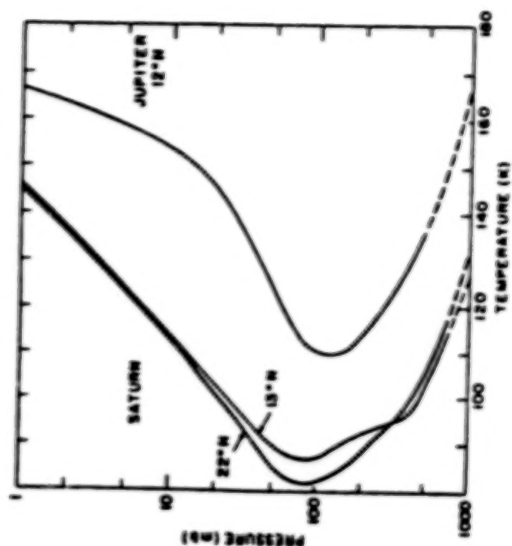


Figure 9. - Saturn environmental models (refs. 21 to 23).

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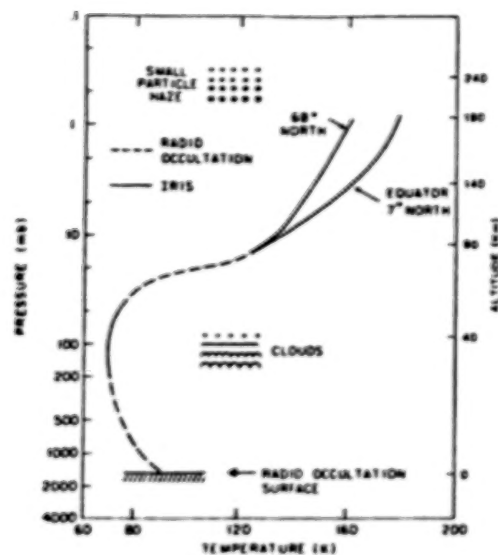
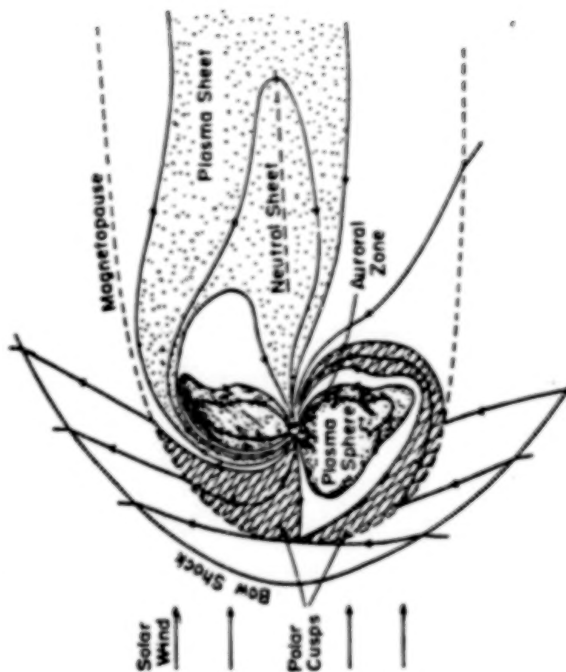
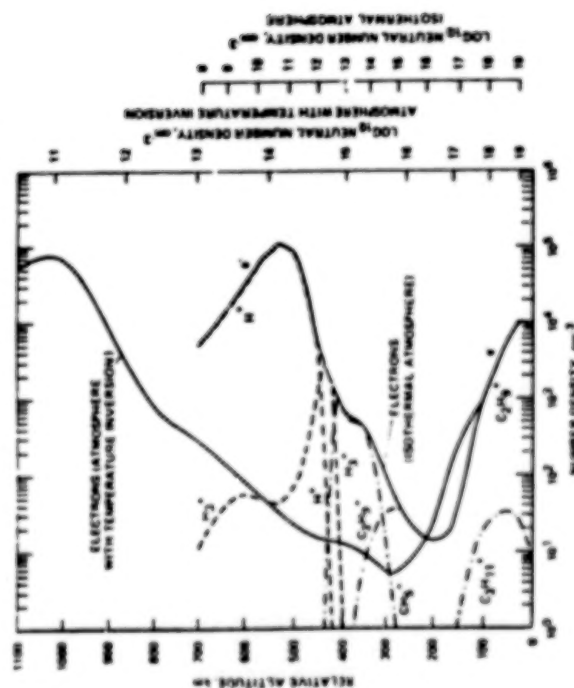


Figure 10. - Vertical temperature profile of Titan's atmosphere (ref. 24).

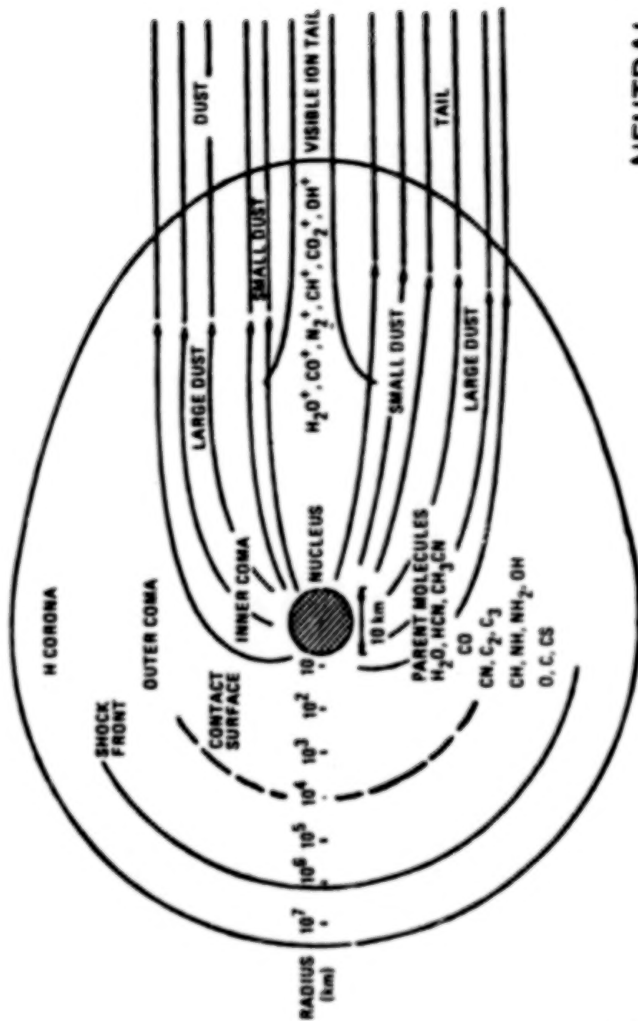
MAGNETOSPHERE (HYPOTHETICAL)



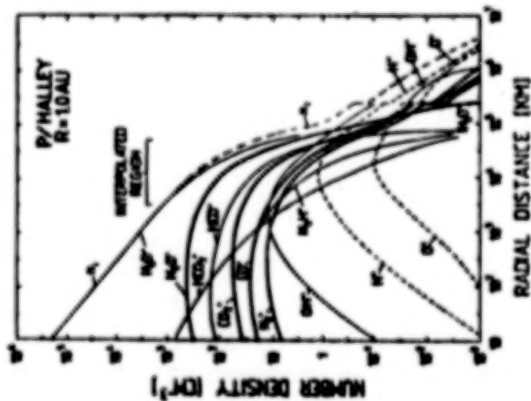
IONOSPHERE (HYPOTHETICAL)



PRINCIPLE FEATURES



IONOSPHERE



NEUTRAL ATMOSPHERE

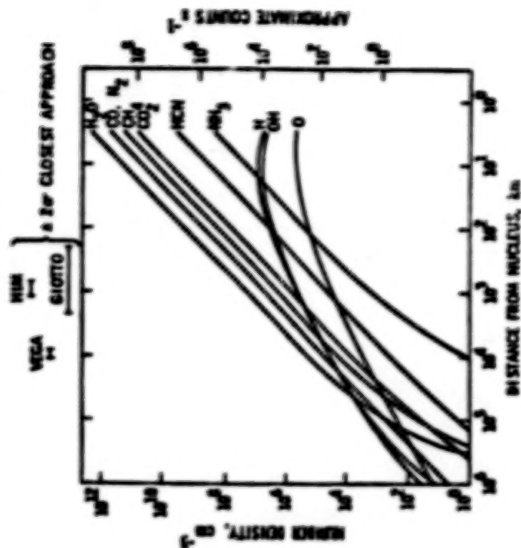


Figure 12a. - Comet environmental models (refs. 28 to 30).



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Figure 12b. - Potential contours around a spacecraft during cometary encounter (ref. 30).

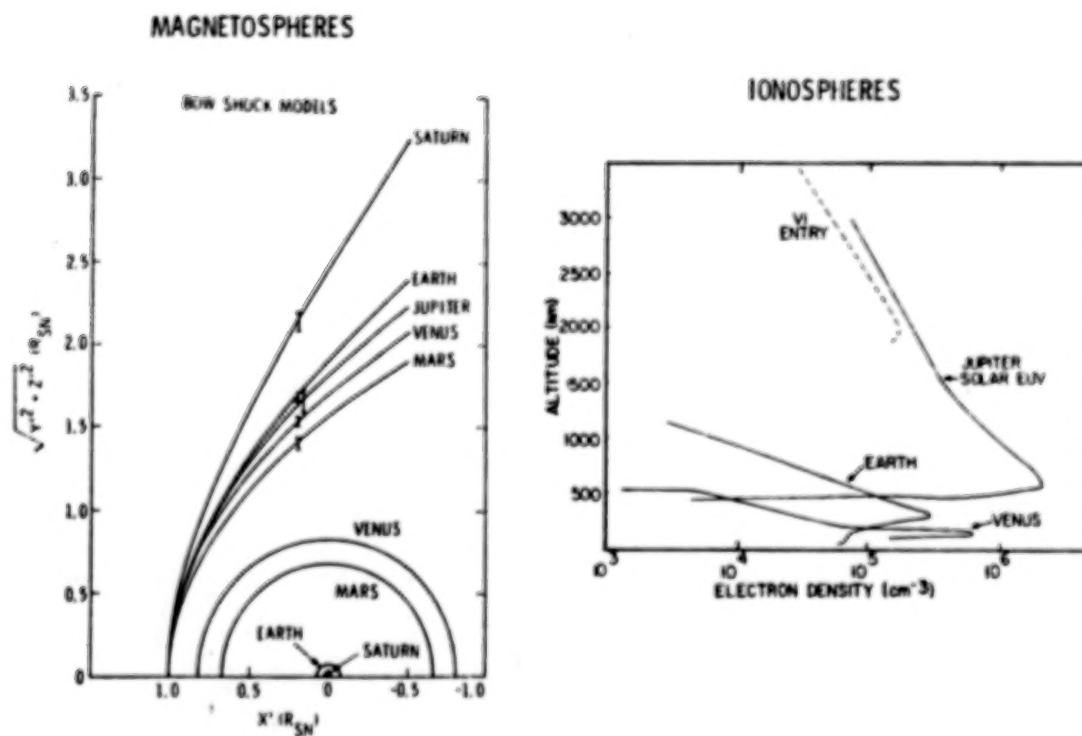


Figure 13. - Comparison of planetary magnetospheres and ionospheres (ref. 31).

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TABLE I.

ESTIMATED PLASMA PARAMETERS/POTENTIALS IN THE SOLAR SYSTEM

REGION				CHARAC- TERISTIC ENERGY, eV		λ_p , m		j_{sw} , V, km/s nA cm ⁻²		POTENTIAL, ϕ V					
										SUNLIGHT			ECLIPSE		
				I ⁺	E ⁻	I ⁺	E ⁻	I-D	RAM	S-D	I-D	RAM	S-D		
VENUS	290 km	10 ⁶	O ⁺ , O ₂ ⁺	0.05	0.3	0.006	0.01	8	8	-1.2	-1.05	-0.85	-1.8	-1.2	-0.80
	1500 km	10 ²	O ⁺	0.2	1	0.33	0.74	8	8	6.0	6.05	2.4	-6.8	-4.45	-2.8
EARTH	150 km	10 ¹⁰ D ⁺	O ⁺ , O ₂ ⁺ , NO ⁺	0.1	0.2	0.007	0.01	8	2	-1.1	-0.75	-0.55			
		10 ¹⁰ N ⁺	NO ⁺	0.05	0.1	0.05	0.07	8	2				-0.58	-0.335	-0.37
	1000 km	10 ¹⁰ D	O ⁺	0.3	0.4	0.34	0.65	8	2	-1.3	-1.25	-1.2			
		10 ¹⁰ N	H ⁺	0.2	0.2	0.23	0.23	8	2				-0.75	-0.725	-0.72
	3.5 R _E	10 ³	H ⁺	1	1	0.23	0.23	3.7	2	-1.85	-1.8	-1.4	-3.85	-3.2	-2.5
GEOSYNCHRONOUS	5.82 R _E	2	H ⁺	5000	2500	370	260	3	2	2.0	1.8	2.05	-6500	-2300	-65005
HIGH LATITUDE	0.1		H ⁺	200	200	130	130	800	2	15	15.5	15	-750	-4005	-500
JUPITER															
COLD TORUS	3.5-5.5 R _J	50-1000	S ⁺ , O ⁺ , O ⁺⁺	0.5	0.5	0.74	0.74	44	0.08	-0.75	-0.585	-0.72	-2.3	-1.35	-1.8
				2	1	0.33	0.23	80	0.08	-3.8	-2.25	-3.1	-4.2	-2.35	-3.3
HOT TORUS	8.0-8.5 R _J	1000-100	S ⁺ , O ⁺⁺	40	10	1.5	0.74	75	0.08	37	-545	-33	-38	-545	-33
				80	20	6.8	3.3	100	0.08	45	-605	-60	-78	-705	-65
PLASMA SHEET	8.0-15 R _J	12	H ⁺ , S ⁺⁺	50	50	15	15	150	0.08	110	-110	945	-180	-170	-1305
OUTER MAGNETOSPHERE	0.01		H ⁺	1000	1000	2300	2300	250	0.08	8.8	8.5	8.55	3800	-4400	-25005
SOLAR WIND	0.3 AU	50	H ⁺	40	85	8.8	8.5	500	20	4.8	4.95	4.4	-260	-1505	-180
	1.0 AU	2	H ⁺	10	50	17	37	450	2	7.8	8.0	7.35	-230	-120	-1105
	5.2 AU	0.2	H ⁺	1	10	17	53	400	0.08	7.4	8.0	8.05	-50	-18	-215

MOST VALUES ARE ROUGH ESTIMATES (SEE APPENDIX B)
 * D MEANS DAY, AND N NIGHT
 † SEE APPENDIX B FOR DESCRIPTION OF COMPUTATION AND CAPTIONS
 ‡ 'PREFERRED' ESTIMATES

TABLE II.

IONOSPHERE

Upper limits:

4000 electrons/cm³ nightside1500 electrons/cm³ dayside.

NEUTRAL ATMOSPHERE

Possible Constituent	Subsolar Point Density (atoms/cm ³)	Partial Pressure (mbar)
He	4500	5×10^{-13}
H	8 (thermal component) 82 (non-thermal component)	
Ne		
Ar		$< 1.2 \times 10^{-10}$
O		$< 6.9 \times 10^{-11}$
C		$< 5.6 \times 10^{-13}$
		$< 1.5 \times 10^{-14}$

TABLE III.

CONSTITUENTS DETECTED IN THE VICINITY OF IO AND IN THE ASSOCIATED PLASMA TORUS

Experiments	Measurement region	Constituent	Density (cm ⁻³)	Temperature (K)
Ground-based telescopes	Cloud around Io	Na	10	$T_i \sim 10^4$
	Cloud around Io	S ⁺		
Pioneer 10 UV experiment	Incomplete torus at Io's orbit	H		
Pioneer 10 radio occultation experiment	Pre-sunset ionosphere	e	6×10^4	
	Pre-sunrise ionosphere	e	9×10^3	
Voyager UV experiment	Plasma torus remote sensing	S ²⁺	95	$T_i \sim 10^4$
		S ³⁺	55	
		O ²⁺	850	
Voyager plasma science experiment	Plasma torus $5.3 R_J < r < 6.3 R_J$ in situ	O ²⁺	Total ~ 2,000	$T_i \sim 10^4$
		S ²⁺ or O ⁺		
		S ⁺		
		S ₂ or SO ₂		
Voyager planetary radio astronomy experiment	Plasma torus in situ	e	2,000-4,000	$T_i \sim 10^4$
Voyager imaging experiment	Volcanoes	Dust umbrella		
Voyager IR experiment	A volcanic plume	SO ₂		

TABLE IV.

PHYSICAL DATA

OBJECT	EQUATORIAL RADIUS [km]	MASS [kg]	DIPOLE MAG MOMENT [G-cm ³]	$\vec{E} = (10^{-8}) (\vec{V}_s \times \vec{B})$ ↓ INDUCED E-FIELD AT SURFACE [V/cm]
SUN	6.960×10^5	1.991×10^{30}	$\sim 3.4 \times 10^{32}$	$\sim 4.4 \times 10^{-1}$
MERCURY	2.439×10^3	3.303×10^{23}	5×10^{22}	1.0×10^{-5}
VENUS	6.050×10^3	4.870×10^{24}	~ 0	—
EARTH	6.378×10^3	5.976×10^{24}	8.1×10^{25}	2.5×10^{-3}
MOON	1.738×10^3	7.353×10^{22}	0	—
MARS	3.398×10^3	6.421×10^{23}	$< 10^{22}$	$< 10^{-6}$
JUPITER	7.14×10^4	1.899×10^{27}	1.59×10^{30}	1.84×10^{-1}
IO	1.82×10^3	8.916×10^{22}	$\sim 6.5 \times 10^{17}$	$\sim 2 \times 10^{-10}$
SATURN	6×10^4	5.686×10^{26}	4.3×10^{28}	5.0×10^{-3}
TITAN	2.56×10^3	1.359×10^{23}	—	—
URANUS	2.6145×10^4	8.66×10^{25}	$\sim 1.9 \times 10^{28} (?)$	$\sim 1.6 \times 10^{-2} (?)$
COMETS	$\sim 1-10$	$\sim 10^{12}-10^{17}$		

TABLE V.

ATMOSPHERIC COMPOSITION

Gas	Band	Wave Number (cm ⁻¹)	Approximate Mole Fraction
Positively identified:			
Methane (CH ₄)	ν_4	1304	1×10^{-2}
Ethane (C ₂ H ₆)	ν_9	821	2×10^{-5}
Acetylene (C ₂ H ₂)	ν_5	729	3×10^{-6}
Ethylene (C ₂ H ₄)	ν_7	950	1×10^{-6}
Hydrogen cyanide (HCN)	ν_2	712	2×10^{-7}
Tentatively identified:			
Methylacetylene (C ₃ H ₄)	ν_9, ν_{10}	633, 328	—
Propane (C ₃ H ₈)	ν_{26}	748	—

N85

13899

UNCLA

ADVANCED CONCEPTS LIQUID DROPLET RADIATOR

A. Hertzberg
University of Washington

The presentation on the advanced liquid droplet radiator concept as a method for the thermal management of future space power systems was based on the content and details of two papers:

1. "The Liquid Droplet Radiator - An Ultraweight Heat Rejection System for Efficient Energy Conversion in Space," by A. T. Mattick and A. Hertzberg, XXXII Congress, International Astronautical Federation, IAF '81, Rome, Italy, September 6-12, 1981.
2. "Light-Weight Nuclear-Powered OTV Utilizing a Liquid Droplet Radiator," by G. Beals, J. Chin, J. Day, C. Gibson, K. Grevstad, G. Larson and D. Treiber, AIAA-83-1346, AIAA/SAE/ASME, 19th Joint Propulsion Conference, University of Washington, Seattle, WA, June 27-29, 1983.

These papers present a detailed discussion of a light weight advanced space radiator concept with application potential ranging from low temperature to high temperature radiator systems. Application of this concept with an advanced nuclear-powered dynamic power generating system is presented.

The first of these papers deals with the fluid mechanics and thermal physics of the liquid droplet radiator. The second paper deals with exploratory systems integration, utilizing the liquid droplet radiator in connection with a nuclear power system.

Since the discussion on advanced concepts is based on the detail presented in these papers, for the convenience of the readers, these papers are included in their entirety as part of the proceedings.

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A LIGHTWEIGHT NUCLEAR POWERED OTV UTILIZING A LIQUID DROPLET RADIATOR

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Abstract

An exploratory point design study was carried out on a shuttle-launchable megawatt nuclear OTV with a 5000 kg payload capacity. The system, which consists of a fixed bed reactor, a Brayton cycle power conversion system, and a liquid droplet radiator to reject heat, is deployable from a small package. The methods and technologies of this design will be discussed, as well as critical design problems. While this is a preliminary study, it indicates that a space-nuclear reactor, combined with the LDR, make possible a continuous 10 MW_e power station on orbit with a single shuttle launch.

1. Introduction

The space transportation system provides a means of easy access to near-Earth space and its unique zero-gravity environment. The exploitation of this environment through space-based manufacturing holds the potential for important scientific and economic benefits. As the level of activity in space increases in the coming decades, the need for generation of multi-megawatt power on orbit will arise for space manufacturing and other space-based applications. Solar power can in principle be used, but the engineering problems associated with it become more and more complex as its power level increases into the megawatt range. Further, large solar cell arrays will require several launches and on-orbit assembly because systems of such size will not be easily deployable from a single launch package.

An important alternative is nuclear power. The high power density and small volume of a nuclear reactor allow it to be scaled up to megawatt levels more easily than solar power systems. A nuclear power system makes a space radiator necessary to reject waste heat that results from the power conversion cycle. The most advanced radiator systems currently available are arrays of heat pipes or pumped liquid radiators, however, such systems are not easily deployable. In addition, to achieve reasonable energy conversion efficiency, the heat rejection temperature should be low. Since radiator area increases as $1/T_{rej}^4$, the number of heat pipes required, and thus the radiator mass, will become prohibitive at high power levels. Even at high rejection temperatures, the mass of the heat pipe radiator remains a large fraction of the total power system mass.

A new concept under study at the University of Washington called the liquid droplet radiator (LDR) offers the possibility of greatly reduced radiator mass.¹ As opposed to using a solid surface, the LDR uses a sheet of recirculating droplets to radiate heat. The advantages of this system over heat

pipes are that with current space structure technology, the LDR is in principle deployable, and because of its low mass to radiating area ratio, it scales up to megawatt power levels without its mass dominating that of the entire power system. As a first order comparison, high performance heat pipe radiators have a mass to area ratio of about 7 kg/m². For the liquid droplet sheet, the mass to area ratio for lightweight oil (Dow 705) droplets of 25 μ m radius is .007 kg/m², or up to 1000 times lighter than heat pipes. While this estimate neglects the LDR machinery and support structure, it is a striking comparison. It indicates that this system presents an opportunity for using relatively low heat rejection temperatures to obtain high cycle efficiencies and increased system reliability, along with a significant reduction in radiator mass.

As one possible application of the LDR with space-based nuclear power, a design study was performed at the University of Washington to examine the feasibility of a nuclear-powered, shuttle-launchable orbital transfer vehicle (OTV) capable of lifting a 5000 kg payload from low Earth to geosynchronous orbit. The design objectives were a total mass of less than 20,000 kg (the shuttle capacity is about 30,000 kg), and a 7-year refueling life. In order for the spacecraft to be able to fly quickly enough to minimize Van Allen radiation, it was estimated that, with the high spacecraft power density achievable through the use of the LDR, a nuclear power system capability of ~10 MW_e electrical output would be sufficient using current electric propulsion technology.

The main components discussed in this paper are the nuclear reactor, the Brayton cycle power conversion system, the liquid droplet radiator, and the deployable spacecraft structure. These components were studied and then integrated into a baseline spacecraft design, which is summarized in Table 1. The dimensions of the total undeployed spacecraft package, excluding the propulsion system, are 11.8x2.5x2.5m, which can easily fit into the shuttle orbiter bay (17.7x4.4x4.4m). With a net power output of 8.7 MW_e and a total mass of 16,200 kg, the launch mass to power output ratio, an important figure of merit in comparing space power systems, is 1.9 kg/kW. The authors know of no other shuttle launchable megawatt power system, either solar or nuclear, that can achieve this power to mass ratio.

Although the results of this study are only preliminary estimates, they indicate that a space-nuclear reactor, combined with the LDR, make possible a continuous 10 MW_e power station on orbit with a single shuttle launch.

*ATAA student members.

Nuclear Electric Orbital Transfer Vehicle

NUCLEAR POWER SYSTEM COMPONENTS	MASS	COMPONENT CHARACTERISTICS
Fixed bed reactor	2200 kg	Operating temperature 1100K Thermal power output 37 MW _t
Shadow shield	1200 kg	Lithium-hydride/tungsten composite
Brayton cycle conversion system	4000 kg	Electrical power output 8.7 MW _e Operating efficiency = 23.4% Non-regenerative; T _{max} = 1100K; T _{min} = 325K
Liquid droplet radiator	5300 kg	Heat rejection fluid-Dow 705 silicone oil* Radiating area = 40,000 m ² Emissivity = 0.9, optical depth = 2.0 Radiating Temperature ΔT = 344-260K
Spacecraft structure	3500 kg	Deployable Astronaut concept Launch package dimensions 11.8×2.5×2.5m
MASS OF NUCLEAR POWER SYSTEM	16,200 kg	
SECONDARY SYSTEM COMPONENTS		
Electric propulsion system and support	1000 kg	8.7 MW _e power required Specific impulse = 2000 seconds
MASS OF ORBITAL TRANSFER VEHICLE	17,200 kg	Within shuttle mass and volume limits

SYSTEM ADVANTAGES

8.7 MW_e power output.

LDR dramatically reduces system mass and increases maximum power output of system.

Fixed Bed Reactor offers a practical, high power density energy source.

Structural design makes the entire system shuttle deployable.

*Trade name of the Dow Chemical Corp.

II. Nuclear Reactor

Space Power Advanced Reactor (SPAR)

Two types of nuclear reactor were considered for the OTV. One is the Space Power Advanced Reactor (SPAR) under study at Los Alamos National Laboratory.² The SP-100 is a reactor of this type, and is presently intended to produce 100 kW_e. The SPAR is a fast-spectrum, uranium dioxide reactor designed to produce 1.2 MW_e. Its cylindrical fuel core (310×390 mm) is contained in a molybdenum can 2.4m thick, and heat is extracted by concentric rings of heat pipes running through the core. The reactor rate of fission is controlled by rotatable beryllium drums with B₄C absorbing segments.

Scaling of this reactor to power levels on the order of 50 MW_e appears feasible in principle. As the reactor power increases, so does the necessary number of heat pipes. Los Alamos has established a requirement of 10 m² of heat pipe vapor area per kW of reactor power. Consequently, as the system power level is increased, the reactor size must increase to accommodate a greater number of heat pipes, as well as an increased fuel core volume to maintain criticality. For a core height to diameter ratio of 1, a UO₂ volume fraction of about .62 is required. Increasing reactor circumference also necessitates more control drums. Through this volumetric analysis, as well as data relating the densities of the reactor components, it is possible to calculate the mass of a scaled up SPAR (Fig. 1).

Fixed Bed Reactor (FBR)

A convectively cooled reactor concept was also examined, one example of which is the gas-cooled fixed bed reactor (FBR) under study at Brookhaven National Laboratory (Fig. 2). The FBR in particular was considered because it was the only reactor of this type designed for space application which had a relatively extensive data base available to the design team. Other types, such as liquid metal cooled reactors, might also be used. The FBR uses

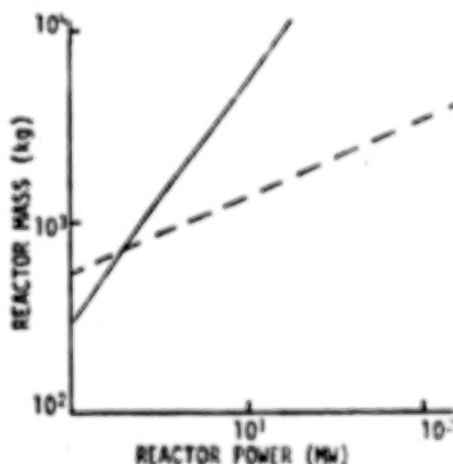


Fig. 1. Mass Scaling Curves for the SPAR and FBR Power Plants.

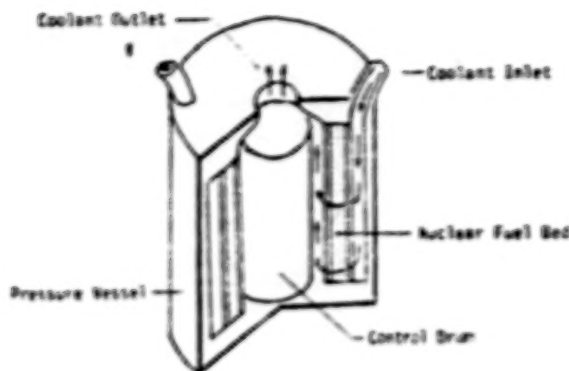


Fig. 2. Fixed Bed Reactor (FBR).

a porous graphite, cylindrical core moderator surrounded by a thin, porous fuel bed made of 400 micron diameter UO_2 pellets. This is enclosed by a beryllium neutron reflector and an aluminum pressure vessel. To cool the reactor, an inert gas mixture at 100 atm is forced through the core and heated to 1100K, while incurring a total pressure loss of about 3%.³ The rate of fission is regulated by the center control drum. The core power density for the FBR is about 500 MW/m³.

Scaling of the FBR is fundamentally different from that of the SPAR because the FBR power level is not volume-dependent. A 500 kg fuel ejection safety cask does add a fixed system mass, however, it is used in remote refueling to lengthen the OTV refueling life. Fuel particles would be pneumatically transferred to the cask, which would be launched into an escape orbit. Fresh fuel would then be transferred into the reactor and operation resumed. This scheme would allow an OTV refueling life of 6 years, or twice the FBR fuel core life of 3 years. Excluding the safety cask, the mass of a scaled-up FBR is as shown (Fig. 1).

Comparing the masses of the two reactors in Fig. 1, the SPAR increases in mass far more quickly than the FBR. This is primarily due to the SPAR heat pipe volume constraint; that is, the volume required to accommodate the heat pipes dominates the volume and mass calculations at high power levels. In fact, the reactor approaches the lifting capacity of the shuttle at around 30 MW_t. Above 2 MW_t, the FBR has a significant mass advantage over the SPAR. Another advantage of the FBR is that the fuel particles it uses have been demonstrated to exceed a 50% burnup. The fuel burnup capability of the SPAR is not certain, but will probably be less than 10%.⁴

Reactor Shielding

As a consequence of choosing nuclear power for this OTV, shielding becomes necessary to protect the crew and equipment from radiation damage. A lithium hydride-tungsten composite shadow shield was chosen as opposed to a complete α shield, which would have made the spacecraft too heavy to be shuttle-launchable. Since the shield thickness requirements for both reactors are equal, the shield mass of the FBR is less than that of the SPAR because the FBR is more compact.

A shield thickness of 0.8m appears sufficient to protect the power system components adjacent to the reactor. Both distance from the reactor and Van Allen belt shielding will further attenuate the dosage to crew compartments and electronics. Detailed calculations will be required to precisely determine the shield thickness, and a somewhat greater thickness may be necessary, however the mass of the additional thickness will be small compared to the total OTV mass.

As a result of this analysis, the fixed bed reactor was selected for this OTV design because of reactor and shadow shield mass advantages at power levels above 2 MW_t, and to avoid the added complexity of heat pipes. As a design choice based on space shuttle payload mass and volume constraints, a reactor power level of 37 MW_t was chosen. Consequently, the FBR has a length of 1.2m, a diameter of 1.1m, and a mass of 2200 kg. The shadow shield is 0.8m thick and has a mass of 1200 kg.

III. Power Conversion System

Brayton Cycle

Dynamic power conversion systems were considered for the OTV because of their advantages of high power density and efficiency over thermoelectric systems at high power levels. After a preliminary examination of both Rankine and Brayton cycles, the Brayton cycle was selected due to its ability to directly use the the inert gas FBR coolant as a working fluid, thus eliminating the need of an intermediate heat exchanger, and also because of its large data base and wide use in present day application. The regenerative Brayton cycle evaluated is modeled after the nuclear Brayton unit developed by AikResearch-NASA Lewis, which has undergone 30,000 hours of continuous operation.⁵

An inert gas mixture is used as a cycle working fluid not only because it is the reactor coolant, but also because it is not chemically reactive under a neutron flux, which minimizes corrosion of turbine and compressor blades. Furthermore, it does not present component materials degradation problems due to radiation. A mixture of helium and a heavier inert gas such as xenon has both the good heat transfer and acoustic properties of a low molecular weight gas and also high molecular weight gas characteristics to allow cycle ducting and turbomachinery to be made smaller. Studies have shown that optimal heat transfer characteristics are achieved with a fluid molecular weight of about 30.⁶

Cycle Analysis

A thermodynamic cycle analysis was performed to determine the Brayton cycle parameters. A limiting turbine inlet temperature of 1100K was chosen so as not to exceed current materials technology limits, and a maximum system pressure of approximately 100 atm was allowed. Heat rejection temperatures were constrained by the 250-350K temperature limits of the LDR fluid to keep the fluid from evaporating or becoming too viscous to pump. Compressor and turbine efficiencies were taken to be .84 and .86 respectively, and are conservative based on current turbomachinery technology. Total

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pressure losses of 3% were estimated through the reactor core and heat rejection heat exchanger.

The primary design constraints were that the LDR have the capability to reject all the heat produced by the reactor in case of an electrical generator shutdown. From this information, a maximum power production capability to total spacecraft mass ratio could be calculated. First, the Brayton cycle rejection temperatures were matched to the heat rejection temperatures of the LDR, and then the resulting cycle efficiency and spacecraft reactor, cycle component, and radiator masses were calculated while independently varying the compressor pressure ratio, rejection temperature, and LDR sheet droplet density.

The Brayton cycle parameters that gave the highest power to mass ratio were determined, and the power level of the spacecraft was scaled up such that the undeployed spacecraft size and mass approached the shuttle capacity, with some margin for error. Through this analysis, the effect of the regenerator was found to be so small that it was decided to remove this component completely. Regeneration in space Brayton cycles is normally used due to the high heat rejection temperatures necessary to minimize the mass of conventional heat pipe radiators. At the relatively low rejection temperatures possible using the LDR, however, the thermodynamic advantage of the regenerator is not sufficient to warrant its inclusion. This appears to be a particularly significant finding, since the mass and volume of the regenerator tend to dominate the mass and volume of a space power conversion system. In addition, the complexity of the regenerator causes it to be failure prone. Tests carried out by NASA Lewis on a space Brayton cycle show that the regenerator was consistently the point of failure of the system.⁵ A small but acceptable decrease in cycle efficiency (from 26.1% to 23.4%) avoided this severe reliability penalty. For a reactor output of 37 MW_t, this efficiency yields a total system power capability of 8.7 MW_e. The design Brayton cycle parameters are as follows: compressor inlet temperature = 325K, compressor outlet temperature = 650K, turbine inlet temperature = 1100K, turbine outlet temperature = 650K, turbine inlet pressure = 1500 psi, compressor pressure ratio = 4.6, turbine pressure ratio = 0.23.

The Brayton cycle turbomachinery mass was estimated by scaling the shaft power of aircraft turboshaft engines, for example the Detroit Diesel Allison T56. This does not suggest that this engine actually be used in the system. It is used for weight and volume estimates only. If three T56 engines are run in parallel so that their combined design power output is near the system electrical power output, their masses total 1700 kg. The length of these engines is approximately 2.5m. High speed alternators are coupled directly to each Brayton unit for added system redundancy. One type of alternator now under study for megawatt power application measures approximately 20x60 cm and has a mass of 230 kg.⁴ The mass of the entire power conversion system, including heat rejection heat exchanger and the three Brayton-alternator combinations, is estimated to be 4000 kg.

IV. Liquid Droplet Radiator

Radiator Description

The advantages of deployability and a low mass to radiating area ratio make the liquid droplet radiator (LDR) particularly effective when combined with a high energy density nuclear reactor. The resultant high power to mass ratio of the power system reflects in improved system performance, greater effective payload, and provides an option for megawatt heat rejection in space.

A typical LDR configuration is shown in Fig. 3. Heated fluid is ejected by generators on one end in a converging sheet of droplets toward a collector on the opposite end, rejecting heat as they cool. The fluid is then ejected back in the opposite direction in a continuous cycle. This cyclic arrangement alleviates the necessity of piping to return the fluid back to the heat source from the first collector. The droplets are generated by a pressurized liquid reservoir with an array of small holes in one side, similar to the process current IBM ink-jet printers employ.⁷ The ability to produce uniform droplets of sizes consistent with this LDR application using fluids appropriate for the system has been validated by Mattick. By etching silicone plate, uniform holes in a liquid droplet generator can be made as small as 35 μ m in diameter, which produce 50 μ m diameter droplets. The aiming accuracy of such holes approaches ± 0.002 rad, which, for a 4m diameter target droplet collector, allows a maximum droplet stream length of 1000m. This limit might be shortened to keep droplet streams from colliding, although this aiming accuracy is considered conservative. While at large distances adjacent droplet streams may collide due to aiming errors, the collision velocity will be small compared to the droplet velocity, and the resultant fluid loss from collisions is expected to be negligible. Droplet coalescence due to collisions will also change the radiative characteristics of the droplet sheet, but at the distance downstream that the droplets will have collided, they will have already radiated most of their heat away. An extra margin in sheet area has been allowed for this. The droplet generator banks themselves will be individually articulated to control droplet aiming during OTV maneuvers and also to correct for structural deflections and vibrations. Several schemes for droplet collection

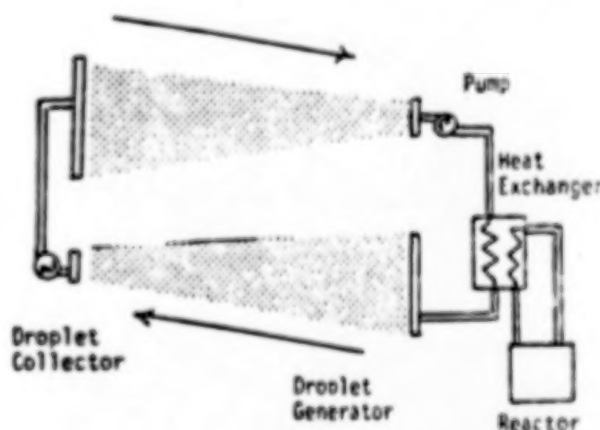


Fig. 3. Liquid Droplet Radiator Schematic.

have been proposed. The one shown is a rotating drum which catches the droplets and reforms them into a continuum by centrifugal acceleration (Fig. 4). Stationary scoops hydrostatically pump the fluid out of the collector.

Radiator Fluids

There are several properties to consider in selecting an LDR fluid, but the most crucial are the fluid vapor pressure and melting temperature. High vapor pressure may lead to significant evaporation and the necessity of constant fluid replenishment. High melting temperature may cause problems in fluid transport through pipes. For a fluid mass loss of less than 20% of the droplet sheet mass over a 5-year period, a vapor pressure limit of 10^{-7} mmHg has been established.¹ An important figure of merit is the radiator power to mass ratio, which is increased when low density fluids of high emissivity (ϵ_0) are used. Commercially available vacuum silicone oils appear quite suitable in all these respects for low temperature heat rejection, i.e., 250-350K, which is the range desired for this nuclear-Brayton system. Silicone oils also have a relatively low density (0.8 g/cc^3). Since solar and Earth radiation produce an influx of heat to the droplet sheet, it would be advantageous to use a fluid which was transparent to visible wavelengths, yet emissive in the infrared band for good heat rejection capability. It turns out that vacuum silicone oils possess nearly precisely these characteristics. Spectroscopic analysis of Dow 705 oil agrees well with published data⁶ and indicate that fluid

emissivity could be as high as $\epsilon_0 = 0.8$. This fluid was obtained through a review of commercially available oils only. Better characteristics might be realized through special tailoring of fluids for this application.

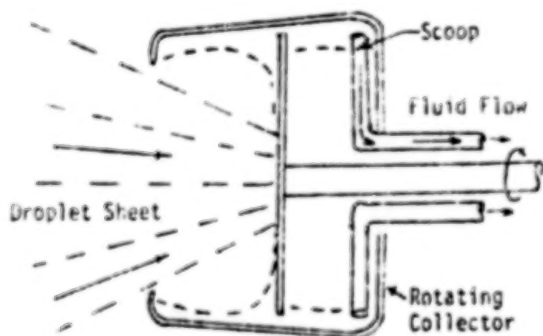
LDR Design and Analysis

The liquid droplet radiator configuration for the OTV is as shown in Fig. 5. A square radiator shape was chosen for simplicity and structural rigidity. Droplets are generated from heated fluid along the side nearest the reactor in a sheet converging to a collector in the lower left corner. From there, fluid is pumped to generators on that same side, which eject droplets to the collector in the upper right corner. The fluid is then pumped to the heat source to complete the cycle.

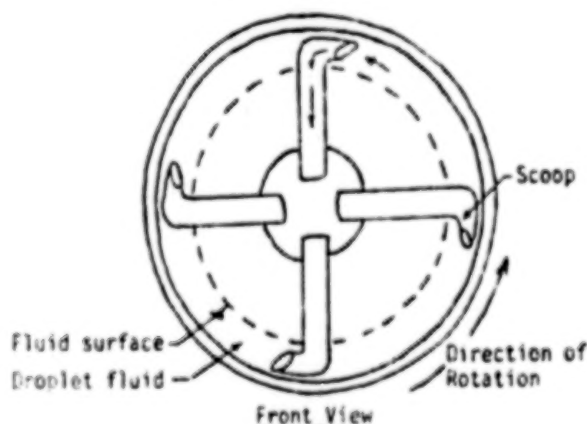
Analyzing the radiating characteristics of a droplet sheet whose geometry, temperature, and droplet density all vary is complicated, but the calculations are eased if some simplifying assumptions are made. The distance traveled by the droplets in flight is taken as the average of the diagonal and straight paths, the sheet temperature is characterized by an equivalent temperature (T_e), which is a constant and a function of the maximum and minimum temperatures of the droplet fluid, and the droplet number density is characterized by an optical depth (τ). The sheet emissivity (ϵ), which is based on the planform area, is calculated from τ .² Although τ varies drastically throughout the sheet, an error of less than 10% results if the value at the center of the sheet is used.

Given the droplet distribution within the sheet, the sheet dimensions, and the maximum and minimum temperatures, important LDR parameters, such as droplet velocity and generator reservoir pressure, can be calculated. Primary design limits are the aforementioned temperature range of the fluid, and a reasonable generator reservoir pressure (<50 atm).

Design selections were again based on maximizing the LDR power to mass ratio. The power to mass ratio of the droplet sheet alone was calculated for various values of τ over a range of T_e (Fig. 6).



Cross Section



Front View

Fig. 4. Droplet Collector.

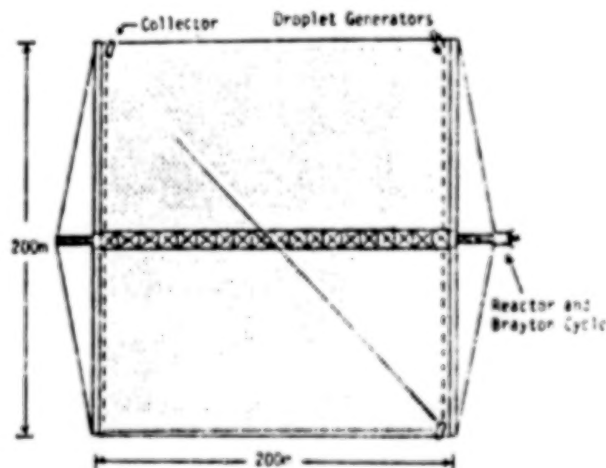


Fig. 5. OTV Planform and LDR Configuration.

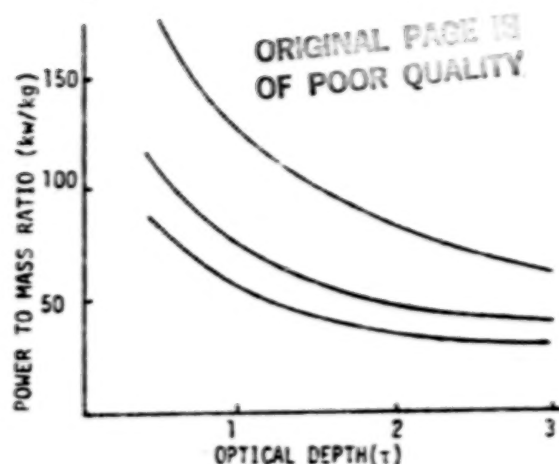


Fig. 6. Variation of Droplet Sheet Power to Mass Ratio with Optical Depth.

For each τ , the corresponding ϵ was used to calculate the sheet power to area ratio for droplet sheet and LDR support structure sizing (Fig. 7). These parameters were then used in the Brayton cycle calculations to determine the maximum power to mass ratio for the entire spacecraft. The main design tradeoff for the LDR can be seen in Figs. 6 and 7. As τ decreases, the sheet power to mass ratio increases due to the effect of lower sheet mass, but since ϵ becomes smaller with decreasing τ , the sheet power to area ratio decreases, increasing the area of the droplet sheet and the LDR structural mass. Another factor is that above optical depths of 4, the sheet emissivity is relatively insensitive to further increases in τ , which rapidly decrease the sheet power to mass ratio.

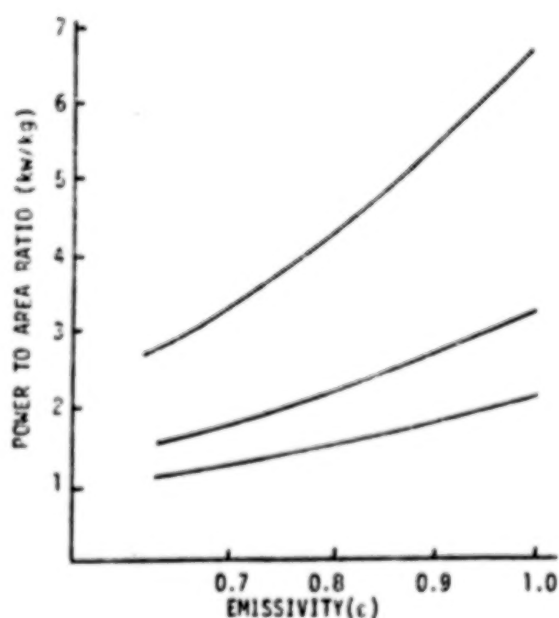


Fig. 7. Variation of Droplet Sheet Power to Area Ratio with Sheet Emissivity.

The Brayton cycle analysis yielded $T_e = 309.3K$ (344K and 280K were maximum and minimum temperatures of the droplet fluid), $\tau=2$, and $\epsilon=0.9$. These parameters allowed a 37 MW reactor output for an LDR sheet area of 40,000m². For 50 μ m droplets in streams spaced 1 mm apart and a streamwise droplet spacing of 6 diameters center to center, the sheet thickness is 30.6 cm. Droplet velocity is 79.1 m/sec, and the generator reservoir pressure is 25 atm. The mass of the droplet sheet multiplied by a factor of 1.8 to account for fluid in generators, pipes, and the heat exchanger gives a total LDR fluid mass of 2100 kg. The total LDR system mass, including generators, collectors, piping, and support structure, is estimated to be 8800 kg. The power to mass ratio of the LDR, which is a good measure of space radiator performance, is 4.2 kW/kg. Compared to heat pipes, which at this operating temperature have a power to mass ratio of about .07 kW/kg, this system offers a significant performance improvement.

V. OTV Structure

The primary function of the OTV structure is to separate the respective LDR generator and collector pairs. It must also remain stable under anticipated thermal and mechanical loads. The structure is composed of a combination of collapsible trusses (Astromasts) conceived by the Astro Research Corporation, which have been used on spacecraft such as Voyager (Fig. 8).⁹ Lightweight composite materials are used in their construction. Lengthwise members (longerons) and the individual members that form the triangular cross section (battens) are made of graphite-epoxy. Kevlar wires are used in between to increase the stiffness of the truss. Most importantly, these masts are easily deployable and very compact in the stowed condition, as the stowed mast length is about 1/50 of the deployed length. The members and wires are connected by rotating pin joints which are self-locking when fully extended, making the mast rigid under compressive and bending loads. This key feature of deployability allows the space radiator area to be relatively large, and also deployable from a small package.

The structural frame for this OTV design (Fig. 9) consists of one main Astromast (200m), 4 smaller masts to accommodate the LDR generators and collectors (100m each), and 4 secondary masts for spacecraft structural support (20m each). A

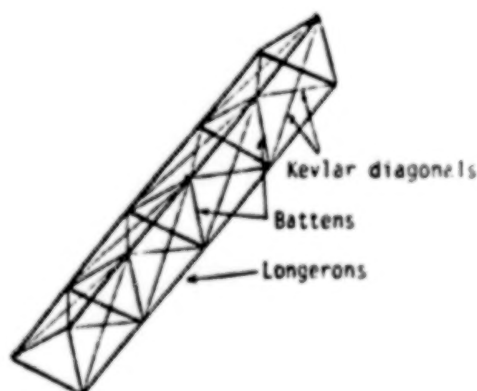


Fig. 8. Astromast Collapsible Truss.

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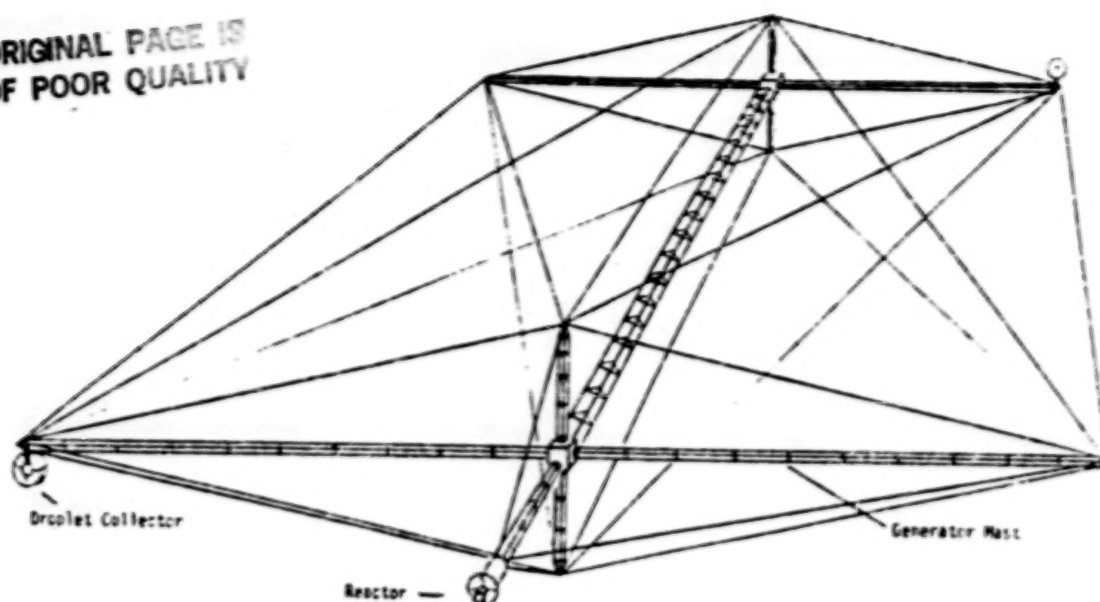


Fig. 9. Nuclear OTV Structural Configuration.

network of Kevlar cables, which are pretensioned for stability, are placed between the masts for support of the spacecraft in bending and torsion. As a rough approximation, the members and wires of these masts were sized by linearly scaling a working model of an Astronaut. The size specifications of each type of mast are given in Table 2. Finite element modelling and analysis performed by D. Treiber show that these estimates are of reasonable order under anticipated loadings and stiffness requirements.

Deployment of the OTV from a launch package can be achieved systematically (Fig. 10). When packaged, the generator and secondary masts are folded onto the main one. This is allowable, since the combined undeployed length of a generator and secondary mast do not exceed that of the primary mast. In the deployment sequence, first the generator and secondary masts swing out 90° and lock into place. The aft end of the spacecraft then rotates 90° about the main mast axis. The main mast deploys, followed by the deployment of

Table 2

OTV Astronaut Specifications

Mast	Mast Diameter (ft.)	Longeron Diameter (in.)	Batten Diameter (in.)	Wire Diameter (in.)
Main	5.57	1.23	1.23	.52
Generator	1.31	.29	.29	.13
Secondary	1.31	.29	.29	.13

An analysis was carried out to investigate structural deflections under design loadings. Only the static loads of the LDR droplet generators and collectors were considered, and it was assumed that these loads are in the plane of the droplet sheet. The masts were modelled as continuous members, and the entire spacecraft frame was analyzed as a 2-D truss. The main design goal was that the in-plane rotations of the droplet generator masts be kept to a minimum, so as to minimize the degree of control input to the generators necessary to keep the droplet streams aligned properly. A Kevlar cable diameter of 1.3 cm for all cables was selected to accomplish this objective.

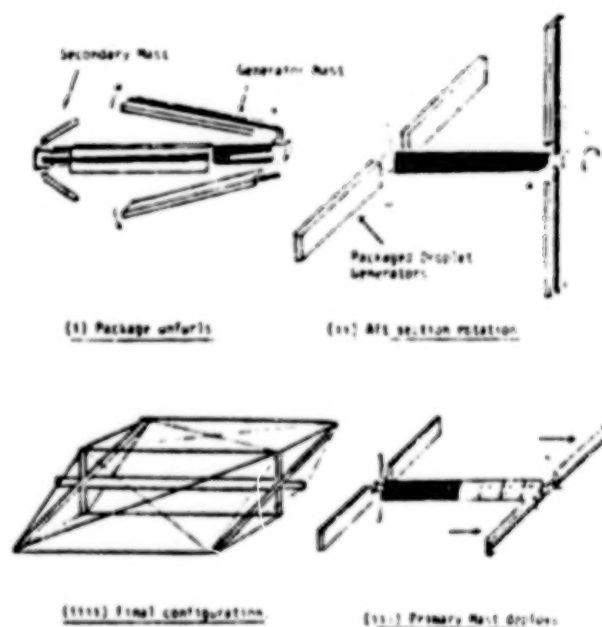


Fig. 10. OTV Deployment Sequence.

the generator and secondary masts. The separate droplet generators, which are originally packaged perpendicular to the generator masts, then rotate 90° into the plane of the droplet sheet (Fig. 11).

From the calculated mast and cable specifications, the total structural mass of the OTV was estimated to be 3500 kg. The dimensions of the undeployed structure, excluding the reactor, shadow shield, power conversion system, and propulsion system, are 6.5x2.5x2.5m using the outlined deployment sequence.

Since this analysis did not consider response to dynamic loads, further investigation is necessary to determine if active structural control is required to dampen various modes of vibration. Damping requirements may, however, be relaxed to only insuring the integrity of the structure itself, as the droplet stream alignment will be independently controlled.

VI. Conclusion

A design investigation was carried out to study the feasibility of a multimegawatt nuclear-powered shuttle-launchable OTV using electric propulsion and a liquid droplet radiator for heat rejection. A 5000 kg payload capacity, a 20,000 kg spacecraft mass, and a 7-year refueling life were the main design objectives.

A 37 MW_e convectively cooled reactor (FBR) was selected for this design because of its mass and volume advantages over the heat pipe reactor above thermal power levels of 2 MW_e.

A Brayton cycle was used as the power conversion system because it can directly use the inert gas reactor coolant as a working fluid, thus

eliminating an intermediate heat exchanger. Furthermore, when coupled with the liquid droplet radiator, low heat rejection temperatures (~300K) can be allowed to obtain a conversion efficiency of 23.4% without using excessively high turbine inlet temperatures. A particularly significant finding is that, as a result of the low rejection temperatures possible, regeneration is not necessary.

The deployability feature and high power to mass ratio of the liquid droplet radiator are used effectively when combined with the high energy density of a nuclear reactor. Using lightweight silicone oils as a radiating fluid, it appears that a radiator heat rejection capacity of 37 MW_e can be realized with a total radiator mass of 8800 kg. The resulting power to mass ratio of the LDR system is 4.2 kW/kg, which is a two-orders of magnitude improvement over heat pipe radiators.

Using Astronast collapsible trusses and a supporting network of Kevlar cables as a structural frame, the liquid droplet radiator is deployable. This OTV design allows a 40,000 m² droplet sheet area to be deployed from a 6.5x2.5x2.5m rectangular package. When the reactor, shield, and power conversion system are attached, the entire launch package is 11.8x2.5x2.5m, which would fit easily into the shuttle orbiter bay.

The entire nuclear power system has an output of 8.7 MW_e and a total system mass of 16,200 kg. This yields a system mass to power ratio of 1.9 kg/kW, which is a value not approached by any other shuttle-launchable megawatt power system concept.

It must be emphasized that this study was carried out as a preliminary point design and is meant only to examine the possibility of megawatt space nuclear power. The performance parameters contained herein are the result of engineering estimates, so this configuration cannot be considered optimal. Nonetheless, the estimates are conservative, the configuration allows ample margin for error in system mass and volume, and the technology demands are relatively modest. The authors believe that a nuclear-Brayton system using the liquid droplet radiator can provide space power capabilities deployable by the shuttle which had previously not been conceivable, and offers an important option for achieving orbital multimegawatt power stations for future space applications.

Acknowledgments

The authors wish to acknowledge the help of Professor A.T. Mattick and the counsel of Professor Abraham Hertzberg, who directed the design study. In preparing this paper, Jim Powell at Brookhaven National Laboratory for information on the FBR, and also the assistance of Greg Anderson, Keith McFall, and Don Morris in preparing the figures.

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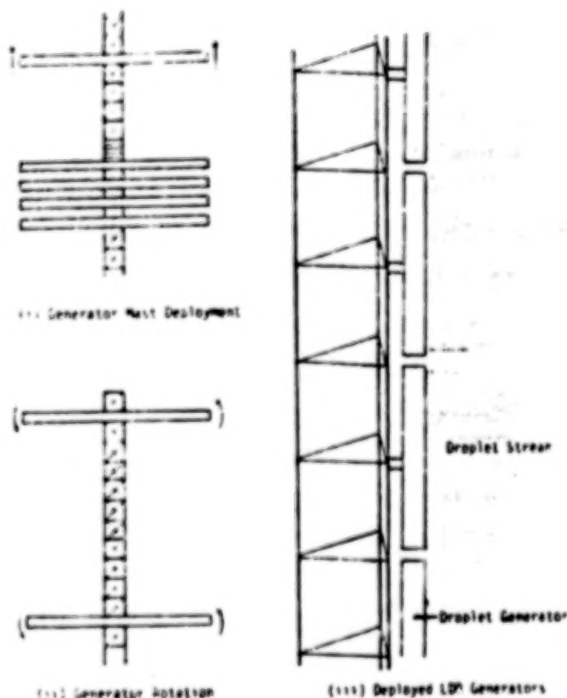


Fig. 11. Droplet Generator Deployment Sequence.

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THE LIQUID DROPLET RADIATOR - AN ULTRALIGHTWEIGHT HEAT REJECTION
SYSTEM FOR EFFICIENT ENERGY CONVERSION IN SPACE

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ABSTRACT

A heat rejection system for space is described which uses a recirculating free stream of liquid droplets in place of a solid surface to radiate waste heat. By using sufficiently small droplets ($<100 \mu\text{m}$ diameter) of low vapor pressure liquids (tin, tin-lead-bismuth eutectics, vacuum oils) the radiating droplet sheet can be made many times lighter than the lightest solid surface radiators (heat pipes). The liquid droplet radiator (LDR) is less vulnerable to damage by micrometeoroids than solid surface radiators, and may be transported into space far more efficiently. Analyses are presented of LDR applications in thermal and photovoltaic energy conversion which indicate that fluid handling components (droplet generator, droplet collector, heat exchanger, and pump) may comprise most of the radiator system mass. Even the unoptimized models employed yield LDR system masses less than heat pipe radiator system masses, and significant improvement is expected using design approaches that incorporate fluid handling components more efficiently. Technical problems (e.g., spacecraft contamination and electrostatic deflection of droplets) unique to this method of heat rejection are discussed and solutions are suggested.

KEYWORDS

Space power; energy conversion; thermal radiation; radiators; emissivity.

INTRODUCTION

A fundamental design constraint for space thermal power systems is the necessity of rejecting heat via radiation. While lower rejection temperatures provide more efficient power conversion and smaller conversion equipment, the radiator size and mass for fixed-temperature radiators increase as $1/T_{\text{rej}}^4$. Moreover, conventional tube and fin radiator designs must incorporate heavy shielding to prevent puncture of coolant tubes by micrometeoroids. Fine or radiation surfaces must be thick enough to assure adequate heat conduction. These requirements dictate high rejection temperatures with correspondingly low conversion efficiencies so that the radiator mass is not unreasonably large.

Recently a heat rejection system for space has been proposed which replaces the solid radiating surfaces of conventional radiators by a recirculating free stream of liquid droplets (Mattick and Hertzberg, 1980, 1981). By taking advantage of the large surface to volume ratio of small droplets, the radiating element of the liquid droplet radiator (LDR) can be made many times lighter than the lightest conventional radiators (heat pipes). This creates the opportunity for using low rejection temperatures to achieve high conversion efficiencies in space power systems.

In this paper the operational characteristics of a liquid droplet radiator based on simple radiator configurations are investigated. Following a description of the LDR components, several issues related to radiator performance will be addressed - the effectiveness of a cloud of droplets as a radiator, choice of radiator media, and vulnerability to micrometeoroids. In order to illustrate the design considerations of a practical droplet radiator, specific space power applications are discussed for which the LDR may prove advantageous. Mass estimates of droplet radiator systems for these applications indicate that because the radiating element itself is quite light, the fluid handling components (droplet generator and collector, heat exchanger, and pump) may account for most of the radiator system mass. To realize the low-mass potential of this type of radiator calls for optimization of droplet generation and collection devices and power conversion systems designed *ab initio* to take advantage of this potential. Following the discussion of space power applications, technical problems unique to this radiator are considered and solutions are suggested.

DROPLET RADIATOR COMPONENTS

The central idea of the LDR is to create a large radiating surface of minimal mass by using the small mass-to-surface area ratio (area specific mass) $\rho r/3$ of small droplets. Ironically it appears that, for higher rejection temperatures at least, heavy liquid metals of low emissivity are the most suitable radiator media, principally due to low evaporation rates. The great advantage of using droplets to achieve large radiating areas is illustrated by the fact that for droplet diameters below 100 μm even a droplet radiator using these liquid metals would require many times less mass than the lightest heat pipe radiators (5-10 kg/m^2) to reject a given thermal power.

A droplet radiator system calls for a means of generating accurately oriented sub-millimeter droplets, which, after radiating waste heat in space, may be efficiently collected, if the low mass advantage of this system is not to be offset by droplet loss. Figure 1 illustrates a possible configuration for implementing the droplet radiator for space solar thermal engines. The liquid, after absorbing waste heat from the power conversion cycle on one module, is formed into a converging sheet of droplets which radiate heat as they travel to the droplet collector on the second module. The cooled liquid is reheated at the second module and projected back to the first module to complete the loop. The use of paired modules eliminates the need for a remotely-deployed collector and return piping, while a converging sheet rather than cylindrical geometry maximizes the effective radiating area of the droplets and minimizes the required collector diameter. This is not necessarily the optimal configuration for this type of radiator and an alternative configuration is described later.

Methods of generating and collecting the droplets are shown in Fig. 2. The generator is a pressurized plenum with an array of nozzles to form liquid jets which break up into droplets via surface tension instability. A vibrator may be used to induce perturbations in the emerging jets to control droplet size and spacing. This scheme is utilized in ink-jet printers to produce accurately aimed streams of droplets $<50 \mu\text{m}$ in diameter at rates up to 10^5 Hz (Kuhn and Meyers, 1979).

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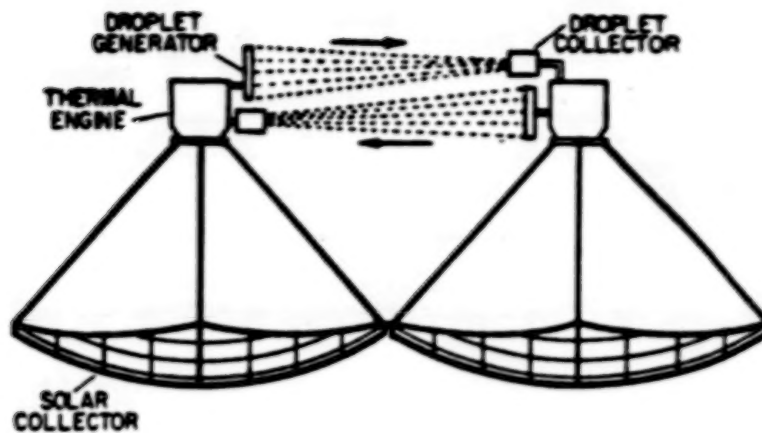


Fig. 1. Solar power satellite using a liquid droplet radiator.

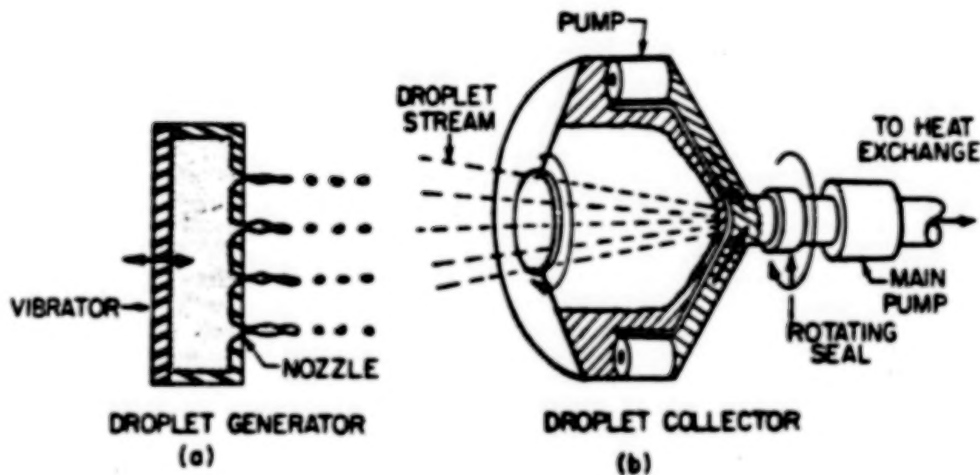


Fig. 2. LDR components, a) generator; b) collector.

The extension of this technique to liquid metals has been under investigation at the University of Washington, and does not appear to present any special difficulties, except for the limited range of containment materials compatible with liquid metals at high temperatures. Experiments on the formation of liquid mercury droplets at room temperature have primarily served to demonstrate the considerable control of droplet size and spacing afforded by acoustically driving the plenum (with a piezoelectric), and to indicate the importance of maintaining the liquid free of impurities. Figure 3 shows photographs of mercury droplet streams (200 μ m droplet diameter) with and without acoustical drive.

The collector must reform the droplets into a continuous liquid under pressure for transfer to a heat exchanger. As shown in Fig. 2b, this may be accomplished by rotating the collector drum so that droplets striking the back surface migrate to the periphery by centrifugal acceleration. Pumps spaced symmetrically about

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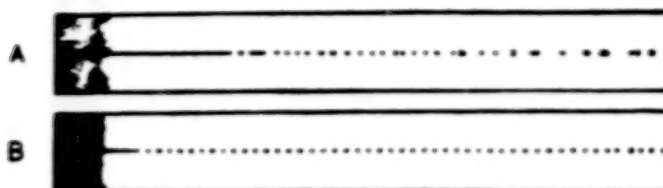


Fig. 3. Mercury droplet streams, generated (a) without acoustic drive, (b) with 9 kHz acoustic drive (droplet diameter = 200 μ m).

the drum periphery are used to overcome the effective head produced by the drum rotation for transfer to the heat exchanger. This rotating vessel scheme has been investigated in connection with metal refining in space (NASA, 1979).

RADIATION BY A SHEET OF DROPLETS

In determining the mass of the liquid droplet radiator for a given heat rejection, account must be taken of the finite view factor of a droplet in the droplet sheet, the intrinsic emissivity of the liquid ϵ_0 , and the decrease in radiation rate as the droplets cool in flight. Although the droplets radiate most effectively when their separation is large, this leads to large radiator areas and an increase in radiator sheet mass must be accepted to realize practical radiator areas. For a dense cloud of droplets it is convenient to characterize its radiative properties by the optical depth $\tau = n\sigma S$ and the hemispherical emissivity of the droplet sheet ϵ , where n is the number density of droplets, $\sigma = \pi r^2$ is the droplet cross section and S is the sheet thickness. With this definition of optical depth, the normal transmission through a sheet of black ($\epsilon_0 = 1$) droplets would be $e^{-\tau}$. Choice of τ for a particular heat rejection requirement involves a compromise between achieving minimum mass (small τ) and minimum radiator area (large τ and maximum ϵ). Figure 4 shows, for a range of intrinsic emissivities, the relative mass, $M^* = 2\tau/\epsilon$, of the radiating sheet vs. the relative area $1/\epsilon$, along with lines of constant optical depth τ . The relative mass is the ratio of the radiator mass for a given ϵ_0 and τ to the minimum possible LDR mass (isolated, black droplets, $\epsilon_0 = 1$, $\tau \rightarrow 0$), and the relative area is the ratio of the area for a given ϵ_0 and τ to that required for an opaque, black radiator ($\epsilon = 1$).

For larger intrinsic emissivities $\epsilon_0 \geq 0.5$, optical depths $\tau \geq 1$ appear to yield the best area-mass compromise. For small intrinsic emissivities, $\epsilon_0 \leq 0.2$, characteristic of liquid metals, larger optical depths $\tau = 1-3$ can be used with relatively less mass penalty than for higher emissivity droplets. Physically, this arises from the fact that for low emissivities, most of the power radiated by a given droplet is reflected rather than absorbed by neighboring droplets. The sheet emissivity ϵ can in fact be several times the intrinsic droplet emissivity ϵ_0 for small ϵ_0 at large optical depths (Mattick and Hertzberg, 1981). For a radiator using liquid tin droplets to reject heat at temperatures of 600-1000°K ($\epsilon_0 = 0.1$), a choice $\tau = 2.5$ would yield a sheet mass = 14.5 times that of a radiator using isolated black droplets (having the same density as tin) and would require area about three times the area of an opaque, black radiator. Despite the large mass penalty paid for the low emissivity of tin, the mass of a radiator using 100 μ m droplets would be 3-6 times lighter than the lightest available solid surface radiators and the droplet sheet can be further reduced in mass in direct proportion to the droplet diameter.

The advantage of using liquids having high intrinsic emissivity is evident from Fig. 4. As liquid metals, except for their low intrinsic emissivities, do appear

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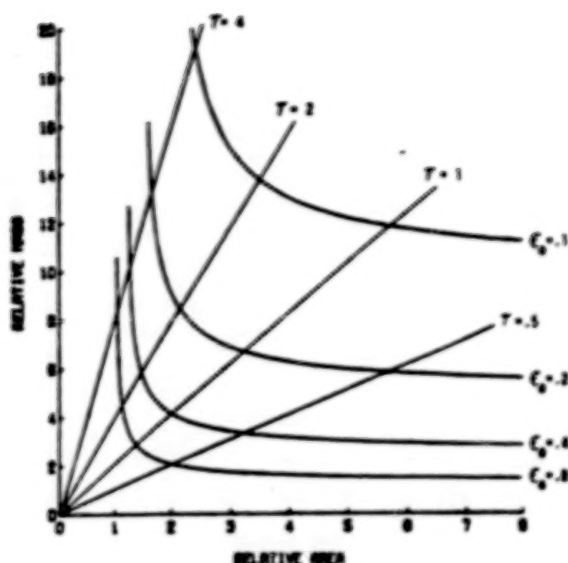


Fig. 4. Relative mass $2\tau/\epsilon$ of radiating droplet sheet vs. relative sheet area for various intrinsic droplet emissivities ϵ_0 (τ = optical depth of sheet).

to be the most suitable radiator media, a means of increasing liquid metal emissivity would be of benefit. A suggested method is to add to the liquid either very small solid particles of high emissivity or an additive in solution which lowers the electrical conductivity of the resulting alloy. It is commonly observed, for example, that an alloy of two metals has a lower conductivity and higher emissivity than either pure metal (Sokolov, 1967). A less obvious method is to utilize droplet sizes comparable to the radiation wavelengths. Mie scattering computations indicate that the absorption cross section exhibits a peak for $\lambda \approx \pi D$ such that the effective emissivity (based on geometric area) is ≈ 2 times the large sphere emissivity at this wavelength. At $\lambda \approx \pi D/10$ the effective emissivity is 0.12 for $\epsilon_0 = 0.1$ and 0.75 for $\epsilon_0 = 0.5$. Utilization of this resonance effect is admittedly best suited to low radiation temperatures where the characteristic wavelengths of radiation are $\geq 10 \mu\text{m}$, since otherwise quite small droplets are required.

While the temperature of the droplets decreases continuously as they radiate energy, it is convenient to characterize the radiation properties of a liquid droplet radiator by an effective radiation temperature T_e such that the power radiated varies as T_e^4 , following the usual thermal radiation law. As shown previously (Mattick and Hertzberg, 1981), the power/mass, a useful figure of merit for a space radiator, can be written as:

$$\text{Power/Mass} = \sigma T_e^4 / \rho_d \langle m^* \rangle \quad (1)$$

where $T_e^4 = 3T_0^4 / (f^3 + f^2 + f)$, $f = T_0/T_1$, T_0 being the temperature at which the droplets are inserted into space and T_1 the collection temperature, and $\langle m^* \rangle$ is the average value of the relative mass over the sheet.

RADIATOR MATERIALS

The most critical property of the liquid chosen for the droplet radiator is vapor pressure. The low-mass advantage of this radiator will, of course, be negated if the mass of fluid required to replenish evaporation losses is much larger than the mass of droplets in the radiating sheet. It has been shown (Mattick and Wertzberg, 1981) that the vapor pressure at the peak droplet temperature should be less than 10^{-7} mm Hg in order that evaporation losses over a 30 year period not exceed the droplet sheet mass. The liquid must also be stable under temperature cycling and radiation exposure, and have good thermal conductivity.

These requirements are met by low vapor pressure liquid metals, notably tin (melting point $T_f = 505^\circ\text{K}$), gallium ($T_f = 330^\circ\text{K}$), and indium ($T_f = 429^\circ\text{K}$), although indium and gallium are probably too rare to be practical. Tin appears to be an excellent medium for high temperature heat rejection since its vapor pressure is less than 10^{-7} mm Hg up to $\sim 1000^\circ\text{K}$, thus affording an operating temperature range of 500°K . For heat rejection at lower temperatures, liquid metal alloys may be used. The tin-lead binary eutectic is usable between $T_f = 456^\circ\text{K}$ and 670°K and the Tin-Lead-Bismuth ternary eutectic between $T_f = 369^\circ\text{K}$ and 550°K . The Na-K eutectic may be used between $T_f = 261^\circ\text{K}$ and $\sim 340^\circ\text{K}$.

There are several oils that have been developed expressly to exhibit low vapor pressures for use as vacuum lubricants and diffusion pump liquids. These oils are typically much lighter than metals and would be suitable for heat rejection near 300°K . Examples are Dow 705 (Pentaphenyltrimethyltrisiloxane) and KEL-F #3 (Chlorotrifluoroethylene). Unlike metals, wherein radiation arises from a thin skin ($\ll \lambda$) at the surface, oils absorb and emit radiation volumetrically (i.e. low reflectivity). Thus, an advantage of the oils is the ability to increase the emissivity to nearly unity by the addition of a suitable dye.

Figure 5 shows the operating temperature ranges for several candidate radiator liquids. The upper temperature for each liquid corresponds to an evaporation loss of $0.03 \text{ kg/m}^2\text{-year}$, taking into account the decrease in temperature of droplets during transit. The lower temperature corresponds to the freezing point in the

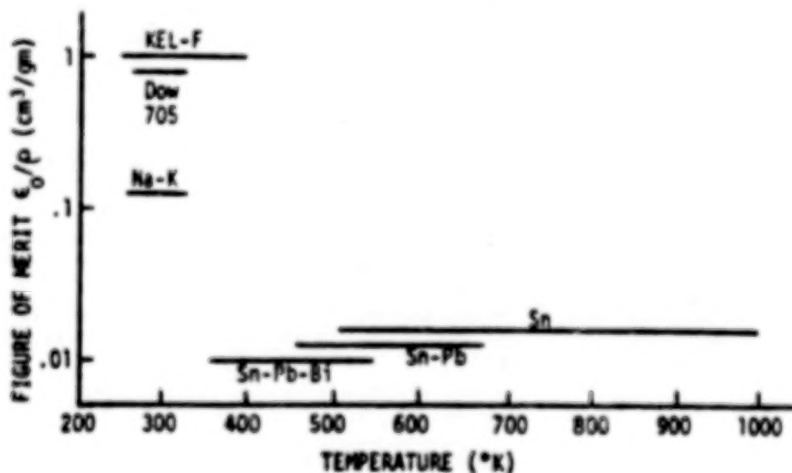


Fig. 5. Operating temperature ranges for candidate radiator fluids. Figure of merit is roughly proportional to power/mass.

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case of metals, and to viscosities >1000 cp for oils. The vertical axis indicates the relative effectiveness of each liquid as a radiator, in terms of the "figure of merit" emissivity/density which, for a given temperature, is roughly proportional to the power/mass. The emissivity of oils is assumed to be $\epsilon_0 = 0.9$ and for metals $\epsilon_0 = 0.1$. Although oils are clearly superior radiator media, so far none have been found which have suitably low vapor pressures above 400°K .

MICROMETEOROID ENCOUNTERS

An advantage of the liquid droplet radiator is the immunity of the radiating surface from single point failure due to micrometeoroid encounters. Solid radiators, by contrast, require massive shielding of coolant tubes to minimize the probability of puncture over the operating lifetime. Although the use of heat pipes reduces the vulnerability of solid surface radiators, puncture of a heat pipe degrades radiator performance to a greater extent than does the passage of a micrometeoroid through a radiating droplet sheet.

To evaluate the effects of micrometeoroid encounters with a droplet radiator, two possible mechanisms of mass loss are considered: droplet deflection and evaporation loss due to heating. In analyzing each mechanism, the micrometeoroids are characterized by a mean speed of 20 km/sec , density $\rho = 2 \text{ g/cm}^3$, and a cumulative flux distribution (Fechtig, 1971) $\phi(m^{-2} - \text{sec}^{-1}) = 10^{-14.6}/m(g)$, so that the probability of a droplet of diameter D being hit by a meteoroid of mass greater than m over a time t is $P = 1 - \exp[-\phi(m) \pi D^2 t]$. Assuming that meteoroids with momenta greater than 10^{-3} times the droplet momentum (with respect to the spacecraft) may deflect droplets sufficiently that they escape, yields for $100 \text{ }\mu\text{m}$ diameter liquid tin droplets travelling at 20 m/sec , $P = 7 \times 10^{-6}/\text{year}$. This is a pessimistic estimate of the fraction of the droplet radiator mass lost due to droplet deflection.

To evaluate evaporative loss due to heating by micrometeoroid impact, a "worst case" assumption is adopted that all of a meteoroid's kinetic energy is channelled into evaporation, irrespective of droplet temperature. With upper and lower limits for meteoroid masses of 1 g and 10^{-20} g , the energy flux of meteoroids is computed to be $2.3 \times 10^{-8} \text{ W/m}^2$, so that the evaporative mass loss due to micrometeoroid heating of a liquid with a vaporization enthalpy of 1000 J/gm would be $7.4 \times 10^{-7} \text{ kg/m}^2\text{-year}$, a negligible fraction of the radiator mass/area.

Another gauge of the effects of micrometeoroid encounters is that a volume fraction of $\approx 3 \times 10^{-5}$ is swept out/year by micrometeoroids with masses greater than 10^{-20} g . Thus, mass loss due to micrometeoroid encounters does not appear to be a significant hazard for the radiating surface (droplet cloud) of a liquid droplet radiator.

APPLICATIONS

The low specific mass of the radiating element of the liquid droplet radiator creates a strong incentive for the design of efficient space power systems which exploit the full advantages of this heat rejection system. In particular, deployment of the LDR involves the use of fluid handling components (generator, collector, heat exchanger pump) which should be integrated with a space power system in a fashion that minimizes system mass. While this design task is a major challenge for the future, some of the factors that must be considered in reaching optimized designs are illustrated by the space power applications discussed below, in which the LDR is incorporated into existing space energy conversion designs as a replacement for solid surface radiators. An extensive space power satellite program conducted by the Boeing Aerospace Company has resulted in detailed designs for thermal and photovoltaic energy conversion and are taken as the reference designs

for LDR applications (Oman and Gregory, 1981). The Boeing thermal conversion systems evolved with the radiator mass and geometry a chief consideration from inception, and the incorporation of a droplet radiator is thus far from optimal, given the unique characteristics of this radiator.

Rankine Cycle

As noted above, the design of thermal engines for space power is largely governed by radiator mass considerations. To keep the mass reasonable, radiator temperatures are high and consequently peak cycle temperatures are at the upper limit of state of the art or extrapolated expander materials. An example is a Rankine cycle heat engine designed by Boeing for a solar power satellite which delivers 10 GW of beamed power to earth. The cycle uses potassium vapor with a peak temperature of 1242°K and a cycle rejection temperature of 932°K and achieves an efficiency of only 18.9%. The heat of condensing potassium vapor is transferred directly to Na-K heat pipes which constitute the radiator. Despite the high rejection temperature and very efficient radiator design, the radiator mass (including fluids) is 15,390 metric tons or 32% of the energy collection and conversion mass. Radiated power for this design is 78 GW, yielding a power specific mass of 0.20 kg/kW_{rad}. The heat pipes alone (radiating surface) constitute 50% of the radiator mass.

In the absence of detailed design studies of liquid droplet radiator components, an accurate mass estimate of a droplet radiator to perform the same heat rejection function as the Boeing heat pipe radiator is not possible. However, a rough estimate is made for this application, principally to elucidate design considerations for the LDR and to indicate the conditions under which the lightweight potential for this radiator may be realized. As the Boeing SPS is divided into 576 individual modules, it is assumed that the specific masses of that design apply to megawatt power levels.

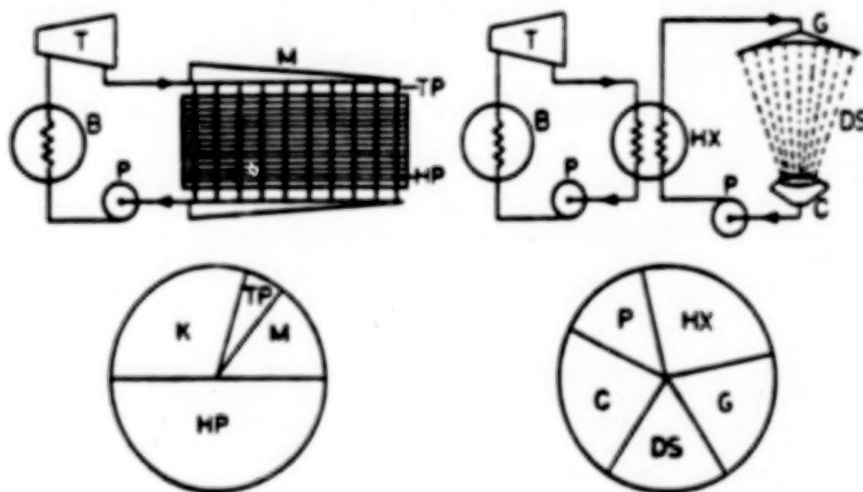


Fig. 6. Potassium Rankine cycle shown with heat pipe radiator and LDR, and mass distribution of each. DS - droplet sheet, K - potassium, M - manifold, TP - throughpipe, HP - heat pipe array, P - pump, HX - heat exchanger, G - generator, C - collector, B - boiler, T - turbine.

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As shown in Fig. 6, the LDR replaces the potassium manifolds, throughpipes, and heat pipes of the Boeing radiator system with a heat exchanger, radiator fluid pump, droplet generator, droplet collector, and radiating droplet sheet. Liquid tin is an appropriate radiator medium with the peak droplet temperature being $T_0 = 932^\circ\text{K}$ and the droplet collection temperature $T_1 = 532^\circ\text{K}$, i.e., a 400°K temperature decrease in flight and an equivalent radiation temperature $T_e = 686^\circ\text{K}$. For a droplet diameter $D = 50\ \mu\text{m}$, optical depth $\tau = 1.5$, and assuming that the intrinsic droplet emissivity can be increased to $\epsilon_0 = 0.2$ by the means suggested above, the relative mass is $m^* = 7.5$, sheet emissivity $\epsilon = 0.4$, and the power specific mass of the radiating sheet is $.033\ \text{kg/kW}_{\text{rad}}$.

The power level of the radiator and masses of the generator and collector are determined largely by the droplet radiation time which, for the chosen parameters, is 3.4 sec. If the pressure in the generator is 1 MPa (pumping power becomes appreciable at higher pressures) the droplet speed $v = \sqrt{2p/\rho} = 17.1\ \text{m/sec}$ so that the sheet length $L = 58\ \text{m}$. Since the droplets can be aimed to within 1 mrad, the collector entrance diameter is taken to be 1 m, which should establish the magnitude of the collector mass. To achieve as low a collector specific mass as possible, a large sheet area is desired, provided that the angular range of droplets does not require enlargement of the collector entrance. The width of the generator is chosen to be 40% of the sheet length, or $W_0 = 24\ \text{m}$, so that the projected area $A = 696\ \text{m}^2$ and the radiated power $P_{\text{rad}} = 7.0\ \text{MW}$. To find the thickness of the sheet, a droplet spacing in the direction of travel of $250\ \mu\text{m}$ (approximately the spacing for free-forming droplets of diameter $D = 50\ \mu\text{m}$) is assumed and the same spacing (between holes) perpendicular to the plane of the sheet. Then, choosing a spacing (at the generator) in the third direction of 2 mm requires a sheet thickness $S = 4.8\ \text{cm}$ to achieve a mean optical depth $\tau = 1.5$.

The generator is a rigid, tapered tube containing liquid tin and having a suitable array of holes. The mass of this component will be almost entirely comprised of the tin and assuming inside dimensions of $24\ \text{m} \times 5\ \text{cm} \times 3\ \text{cm}$ (the 3 cm dimension being an average over the tapered tube length) yields a mass of 245 kg or a specific mass of $.035\ \text{kg/kW}_{\text{rad}}$. The heat exchanger mass is estimated to be 350 kg ($.05\ \text{kg/kW}_{\text{rad}}$), based on the large mean $\Delta T = 200^\circ\text{C}$, and the exceedingly high heat transfer coefficients of liquid tin and condensing potassium vapor. A NaK to sodium heat exchanger designed in 1954 to transfer 1 MW with a $\Delta T = 40^\circ\text{C}$ (Jackson, 1954) had a volume $< 0.014\ \text{m}^3$ for each fluid. Scaled to $\Delta T = 200^\circ\text{C}$ and considering improvements in heat exchanger materials since 1954, $.05\ \text{kg/kW}_{\text{rad}}$ does not appear unreasonable at megawatt levels. For the pump (mass flow rate $67\ \text{kg/sec}$, $\Delta p = 1\ \text{MPa}$, power = 10 kW), a mass of 150 kg ($.021\ \text{kg/kW}_{\text{rad}}$) is estimated.

The collector is likely the most technically complex element of the LDR system. To overcome the centrifugal force of the rotating drum requires either that pumps be mounted on the periphery as shown in Fig. 2 or that a large diameter rotating seal be used. It is conceivable that electrostatic or magnetohydrodynamic means might be employed to collect the liquid instead of the mechanical device suggested. In addition, it would be beneficial if the droplet kinetic energy could be recovered, at least in part, as pressure in the collector. In the absence of a detailed design either for the present concept or for a more advanced collector, a highly speculative mass estimate of 300 kg ($.043\ \text{kg/kW}_{\text{rad}}$) is made for this component (1 meter aperture). A breakdown of radiator mass for both the heat pipe radiator and the LDR is shown in Fig. 6. The mass of potassium is included as a separate entity for the heat pipe radiator system, but for the droplet radiator, the potassium and tin complements are included in the mass of each component.

These estimates yield a specific mass of $0.18\ \text{kg/kW}_{\text{rad}}$ for the liquid droplet radiator in this application, nearly that of the heat pipe radiator. Despite the speculative nature of the estimates, the specific mass is not expected to be more than 50% greater than this. While in this application the LDR does not offer a

large mass advantage over heat pipe systems, it has features that may make it the preferred system. These include its reduced vulnerability to micrometeoroid damage, and the fact that transporting the LDR into space may be far more efficient, since the 700 m^2 radiating surface may be stored as a liquid occupying 0.034 m^3 . In estimating LDR mass for this application, an add-on approach was taken, with no attempt to integrate the LDR efficiently with the Rankine cycle. It is emphasized that the heat rejection "component," the droplet sheet, amounts to only about 1/6 of the total LDR mass. To capitalize on the low specific mass of this component will require careful optimization of fluid handling components and integration of these in the energy conversion cycle.

As mentioned above the most significant advantage of a lightweight radiator such as the LDR is the potential for using low rejection temperatures to improve thermal cycle efficiency. To illustrate this the above example of the use of an LDR in a Rankine cycle is reconsidered, assuming a rejection temperature (peak droplet temperature) of 732°K instead of 932°K . If the same droplet collection temperature $T_c = 532^\circ\text{K}$ is used, the effective radiation temperature $T_e = 619^\circ\text{K}$ and the radiation time $t = 2.58 \text{ sec}$. By choosing a plenum pressure of 1.7 MPa , the sheet length will again be 58 m so that the above dimensions and masses can be taken for the droplet sheet, generator and collector. With twice the mass flow rate, the heat exchanger and pump are conservatively estimated to have twice the mass as above. The power for these conditions is $4.6 \text{ MW}_{\text{rad}}$ so that the power specific mass at the lower rejection temperature is $0.40 \text{ kg/kW}_{\text{rad}}$. Since the heat pipe radiator has a radiating surface at nearly constant temperature near the minimum cycle temperature, the area and mass of such a radiator is expected to increase as $1/T_{\text{rej}}$, i.e., the power specific mass of a heat pipe radiator in this application is estimated to be $0.2 \times (932/732)^2 = 0.53 \text{ kg/kW}_{\text{rad}}$. A cycle redesigned for heat rejection at 732°K would be expected to have an efficiency near 30%, so that the masses of other components could be significantly reduced. Alternatively, a lower peak cycle temperature could be used to reduce thermal stresses.

Brayton Cycle

The liquid droplet radiator may be incorporated with fewer design modifications in Brayton cycles since a heat exchanger is commonly used in the heat rejection system. Earlier studies by Boeing (Woodcock, 1977) in fact concentrated on a helium Brayton cycle design which was later found to be less competitive than the potassium Rankine cycle due to the large radiator mass required to reject heat at a mean radiator temperature of $\approx 500^\circ\text{K}$. The radiator system was comprised of a heat exchanger to transfer energy from helium ($T_{\text{in}} = 685^\circ\text{K}$, $T_{\text{out}} = 404^\circ\text{K}$) to NaK ($T_{\text{in}} = 377^\circ\text{K}$, $T_{\text{out}} = 644^\circ\text{K}$) which was in turn transferred to heat pipes in a manner similar to the Rankine cycle design. The specific mass of the radiator and heat exchanger in this design amounted to $1.4 \text{ kg/kW}_{\text{rad}}$ with heat pipes accounting for about 70% of the radiator system mass. An analysis similar to the above of a liquid droplet radiator using the tin-lead-bismuth eutectic for this application ($20 \mu\text{m}$ droplet diameter) yielded a specific mass of $0.9 \text{ kg/kW}_{\text{rad}}$. The radiating droplet cloud amounted to only 9% of the LDR weight in this case.

Nuclear Power Cycles

Nuclear power cycles appear very promising for space application due to their exceedingly high power density and low mass. Recently a study of a rotating bed reactor (RBR) power cycle for space was carried out (Powell, Botte, and Hertzberg, 1981) showing that thermal power densities as high as 1000 MW/m^3 might be achieved. A compact thermal energy source such as this is suited to LDR designs which maximize the radiator area and may reduce the mass of fluid handling components. An example of such a design (Hayes, 1981) is shown in Fig. 7. In this design, the

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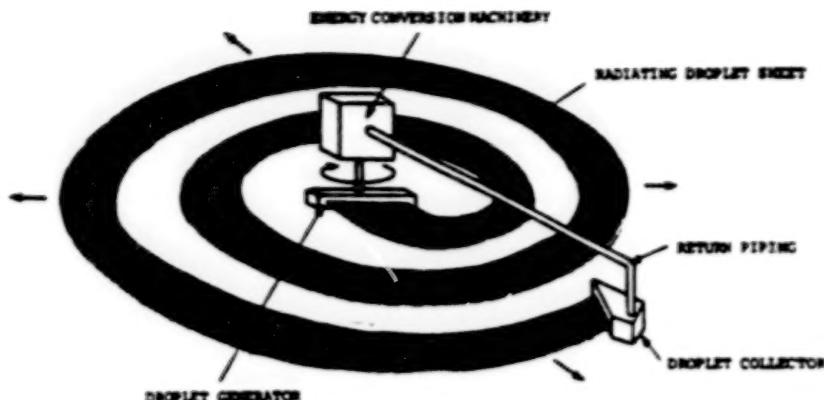


Fig. 7. Rotating-boom liquid droplet radiator.

radiator system rotates as a unit about the satellite center of mass, and droplets, emitted at low speed, follow radial trajectories which results in a spiral radiating sheet. Collection is simplified since the droplets are simply swept up by a collector mounted on a boom, and the centrifugal acceleration at the collection point automatically forms the droplets into a continuous liquid without the need for a rotating vessel. Preliminary analysis indicates that radiator system mass might be reduced by a factor of 2-4 in comparison with the non-rotating LDR.

Photovoltaic Power System

The low mass of the radiating surface of the LDR also creates the potential for improving the performance of photovoltaic conversion systems. Designs for space power using photovoltaics use relatively small concentrations of solar energy since the solar cells are passively cooled and higher concentrations would increase cell temperatures and reduce efficiency. Active cooling would allow solar concentration while maintaining conversion efficiency and may reduce the area of solar cells by factors of several hundred. For an actively cooled system to be competitive with a passively cooled design, however, the masses must be comparable.

In assessing the potential for using the LDR for cooling solar cells in space, the Boeing design (Oman and Gregory, 1981) for a photovoltaic SPS is used. The mass/area of the solar cell array is $.427 \text{ kg/m}^2$ and the cells operate at a mean temperature of 313°K with 17.3% efficiency. Assumed parameters for a silicone oil droplet radiator are a peak droplet temperature $T_0 = 313^\circ\text{K}$ (solar cell temperature), a collection temperature of 263°K ($T_c = 286^\circ\text{K}$), and droplet emissivity $\epsilon_0 = 0.9$. For simplicity it is assumed that at high concentration all of the heat is rejected by the droplet radiator. Since the emissivity of the radiating droplet sheet is nearly that of the panels, and the effective temperature of the sheet is 286°K vs. 313°K for the panels, the area of the sheet must be 1.43 times that of the passively cooled panels. For $50 \text{ } \mu\text{m}$ silicone droplets, with $\tau = 1$, the mass/area is 0.033 kg/m^2 . Thus, each kg of the solar cell blanket is replaced (at high concentration) by .111 kg of radiating silicone fluid. Assuming a concentrator area equal to 1.1 times the initial solar cell area (concentrator reflectivity $R = 0.9$) and a concentrator specific mass of $.07 \text{ kg/m}^2$ (including structure and reflector facets), each kg of initial solar cell mass is also accompanied by .173 kg of concentrator mass. Thus, for masses of actively and passively cooled systems to be comparable, the masses of the auxiliary components of the droplet radiator cannot be greater than 7 times the mass of the radiating surface. This should be attainable in light of

the mass estimates for components of the solar thermal engine radiator described above. The advantage of the droplet radiator, then, is that it considerably reduces the area of silicon cells required, and consequently relieves somewhat the pressure to reduce production costs of these cells.

TECHNICAL CHALLENGES

In addition to the design of low-mass fluid handling components of space power systems, there are several technical challenges unique to the liquid droplet radiator which must be faced in reaching a practical design. Discussed here are those that appear most pressing in terms of spacecraft design.

Contamination of Spacecraft Surfaces and Radiator Fluid

While the evaporation rate of the radiator liquids discussed above is small and not considered to be a large factor in radiator performance, there remains the possibility of contamination of spacecraft surfaces by the condensing vapor. This problem is particularly important for high resolution optics and thermal coatings. As heating contaminated surfaces to vaporize this condensed fluid appears impractical, the best solution is to avoid contamination. In view of the extremely low vapor pressure of the LDR fluids, this may be accomplished by using baffles. Vapor pressures below 10^{-7} mmHg correspond to mean free paths of kilometers, i.e., the range of free molecular flow. Thus, baffling is expected to be completely effective, and should be a consideration in choosing the radiator geometry for each application.

A related problem is the contamination of the radiator fluid by spacecraft effluent such as propulsion gases. The authors' experience has shown that to achieve reliable droplet formation and accurate jet aiming, the liquid must be kept free from at least particulate impurities larger than $\approx 1/10$ the nozzle diameter. Absorbed gases or films on droplets may alter their radiative properties. Again, baffling would minimize this problem. An appropriate solution to this and the evaporation of the radiator liquid may be to enclose the radiator in a plastic film of micron thickness, which will transmit radiated energy in the 2-20 μ m wavelength range.

Droplet Charging

The charging of droplets and the resulting mutual repulsion may cause a significant fraction of the droplets to miss the collector. The charge accumulated by droplets during flight is not sufficient to cause appreciable deflection over 3-10 second transits, but the charge accumulated by the spacecraft over much longer periods may be transferred to the droplets during their formation, which can easily lead to sizeable droplet deflection. This is a consideration at spacecraft potentials as low as tens of volts. Thus, some means of preventing electrostatic deflection is a necessity. This might be accomplished by maintaining a low spacecraft potential using charge ejection, by applying a countering voltage in the region of droplet formation, or perhaps by "guiding" charged droplets with a grid of appropriate potential surrounding the droplet sheet.

Radiator Orientation

Radiator orientation is obviously of major importance if the droplets are to be efficiently collected. Rapid changes in spacecraft attitude could lead to a major loss of fluid for longer droplet transit times. Thus, a guidance and control system must be incorporated which predicts droplet trajectories and positions the

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collector accordingly. While this problem apparently has a straightforward solution, it deserves mention as an essential feature of radiator design. The orientation system and the spacecraft trajectory should insure that the radiator sheet be edge-on to the sun, since the absorptivity of sunlight by liquid metals in particular is typically much higher than the emissivity at temperatures $< 1000^\circ\text{K}$.

Manufacturing of Droplet Generators

High volume fabrication techniques for fashioning droplet generation holes must be developed. One megawatt thermal radiators typically will require 10^5 - 10^6 droplet streams which must be oriented to within 10 mrad, if the collector size is to remain reasonable. The nozzles must be formed in substrates impervious to corrosion by high temperature liquid metals and of sufficient thickness to contain pressures of 10 atmospheres or higher. Although a technology for the formation of accurately aligned micron-sized holes has been developed for ink jet printers, the extension to large arrays and high temperature materials must be developed. Laser drilling, or etching of crystalline substrates, are possible methods. YAG lasers are currently used to drill $60\text{ }\mu\text{m}$ diameter holes in ruby substrates for watch bearings at the rate of 6-10 holes/sec (Nagano, 1978). While vibrating the liquid may not prove necessary, it does provide a convenient control of droplet size and spacing and can facilitate the production of uniform droplet streams. Thus, an efficient means of generating the acoustic drive (10^4 - 10^5 Hz) and coupling to the high temperature fluid bears investigation.

Auxiliary Heating

It is important, for liquid metals at least, that the fluid handling components be provided with heaters to enable system start-up and to avoid freezing in the event of a cycle interruption. During such an interruption, the radiating liquid could be evacuated into an insulated reservoir which would require only a small power to maintain high temperatures. While this consideration applies to any system employing a high fusion temperature material, such as the potassium Rankine cycle described above, it must be included as a necessary contribution to the mass of high temperature LDR's.

CONCLUDING REMARKS

The ultralight weight of the radiating element of the liquid droplet radiator creates the possibility of low-mass, highly efficient energy conversion systems for space. In addition, the use of a cloud of droplets instead of a solid surface to reject waste heat may greatly facilitate transportation of the radiator into space and also reduces the vulnerability of the radiator to micrometeoroid damage.

Since detailed designs for the LDR and other components remain to be developed for specific power conversion applications, only a preliminary assessment of the efficiency of this radiator has been offered. This analysis has shown that the fluid handling components which generate and collect the droplets and transfer heat to the radiating fluid may well account for much of the radiator system mass. To realize the full advantages of this radiator, emphasis must be placed on the design of low-mass fluid handling components, instead of directly substituting the droplet radiator for one of conventional design, and the development and optimization of space power systems must aim from inception to integrate these elements in a manner that minimizes the system mass.

The benefits that may be expected from the space enterprise are tied, no less than on earth, to the ability to provide power economically. The liquid droplet radiator offers a potential for a substantial reduction of mass, and thus cost, of space power generation, and more generally enlarges the scope of applications with a requirement for heat rejection that may be considered for space. While as yet untested in practice, this concept is based largely on existing technologies, and developmental studies of droplet radiator systems are continuing at our laboratories at the University of Washington in cooperation with several other institutions. Many of the questions regarding the droplet radiator's characteristics, however, can only be answered by direct experimentation in space, and it is suggested that an experimental droplet radiator be deployed on the space shuttle in the near future.

ACKNOWLEDGMENT

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ELECTRODYNAMIC TETHERS

I. Power Generator in LEO

II. Thrust for Propulsion and Power Storage

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An electrodynamic tether consists of a long insulated wire in space whose orbital motion cuts across lines of magnetic flux to produce an induced voltage. In typical low orbits this averages about 200 v/km. Such a system should be capable of generating substantial electrical power, at the expense of IXB drag acting on its orbital energy. If a reverse current is driven against the induced voltage, the system should act as a motor producing IXB thrust.

A reference system has been designed, capable of generating 20 KW of power into an electrical load located anywhere along the wire at the expense of 2.6N (20,000 J/sec) drag on the wire. This system consists of 10 km of #2 AWG aluminum wire to provide an average induced voltage of 2 KV, with 10 ampere hollow cathode plasma contacts at each end. In an ideal system, the conversion between mechanical and electrical energy would reach 100% efficiency. In the actual system part of the 20 KW is lost to internal resistance of the wire, plasma and ionosphere, while the drag force is increased by residual air drag. The 20 KW PMG system as designed is estimated to provide 18.7 KW net power to the load at total drag loss of 20.4 KJ/sec, or an overall efficiency of 92%. Orbit reboost propulsion would require 2-4 kg/hr, versus about 8 kg/hr to generate 20 KW using fuel cells.

Similar systems using heavier wire appear capable of producing power levels in excess of 1 Megawatt at voltages of 2-4 KV, with conversion efficiency between mechanical and electrical power better than 95%.

The hollow cathode based system should be readily reversible from generator to motor operation by driving a reverse current using onboard power. This is particularly attractive for application to high drag power systems such as solar arrays, eliminating the normal fuel requirement for orbit maintenance. One kilowatt of (solar array) power converted to electrodynamic thrust should provide the same "delta-V" capability as 1,000 kg/year of fuel expended in OMS/SME class rocket engines. A fully reversible Plasma Motor/Generator system of this type also provides a capability to store excess solar array power output as added orbital energy during sunlight, then tapping that stored energy to provide peak, emergency or night-side power. Projected performance figures are very competitive with batteries or regenerative fuel cells. Power storage capacity for any PMG mounted on a 200,000 kg (space station) is 250 KWHR per kilometer (of allowable altitude change).

All systems discussed in this paper are conceptual, based on theoretical calculations that appear to be well founded but are impossible to confirm without direct measurement using an actual tether wire in orbit at 8 km/sec. The first such tests are now planned for the 1987 TSS-1 (Tethered Satellite System) flight on the Shuttle Orbiter.

INTRODUCTION

If a spaceborne tether system is made electrically conducting, it can provide an electrodynamic coupling between the host spacecraft and the surrounding space magnetoplasma. This coupling could be employed to produce significant levels of electrical power, particularly in LEO, if a satisfactory return path for currents thru the tether wire can be established thru the ionosphere. The wire must be held stable across the magnetic field lines by gravity gradient or centrifugal forces.

The general concept of space tethers has been summarized recently by Bekey(ref.1), and considered in some detail at two NASA Space Tether Workshops (ref.2,3) and studies at the Smithsonian Astrophysical Observatory (ref.4). Applications have been under study at NASA-JSC for several years. The most promising applications identified for the electrodynamic interactions of a conducting tether have been in the area of generation of power and thrust from the VXB induced voltages and the IXB forces on the tether wire. These electrodynamic tether interactions provide a very promising conceptual means for converting mechanical energy (of orbital motion obtained from rocket engines) to electrical energy, or for converting electrical energy (from solar arrays, thermal or nuclear generators) into mechanical energy (of orbital motion obtained without rocket engines or fuel for reaction mass) by reversing the process. The tether wire would be made to serve as the armature of a gigantic orbital electric motor/generator system using the Earth itself as its field/pole pieces and frame, and the ionosphere as its current return circuit.

Estimates of ionospheric conductivity indicate ample margin to carry the required closure currents. Design of 1-1,000 KW tether motor/generator systems should depend primarily on resistive losses in the tether wire itself, losses in coupling of current to the ionosphere at each end, neutral atmosphere form drag, insulation requirements and load impedance/induced voltage matching requirements. Propulsion requirements to offset the IXB drag caused by power generation are estimated to be 100-200 kg per 1,000 KWHR generated, versus about 400 kg consumed using fuel cells. Stability of the tether system, to maintain alignment of the tether wire across the magnetic field lines and perpendicular to the orbital velocity against the disturbing torques due to time variable IXB forces is required.

Stability:

A subsatellite tethered to a massive host spacecraft by a long cable tends to be held in radial alignment by a combination of gravity gradient and centrifugal forces. The tension force in the tether cable due to the subsatellite can be expressed as

$$T = M G(R) 3 L / R$$

where M is the mass of the subsatellite, G(R) is the local force of gravity at the orbital radius R of the host spacecraft center of mass, and L is the length of the tether cable. This tension is relatively small, but sufficient to maintain stable alignment of the tethered system. A 1,000 kg subsatellite tethered at 10 km from a very massive spacecraft in a 230 km earth orbit would produce a tether force of 44 N (9.9 lbs). A light fishing line would be capable of holding this large satellite suspended in position 10 km above or below the orbiting host spacecraft against most disturbing forces.

In particular, if the tether cable used was an insulated wire carrying a current of 10 amperes (20 KW at 2 KV) the IXB force on the wire would produce an in-plane component of 2.6 N. Static stability for this case is satisfactory, a deflection of only 3.4° being caused by this disturbing force. In fact, for most

motor/generator concepts considered, the mass of the tether wire alone provides sufficient force to stabilize the system with no satellite mass at all. For such a massive tether system the tension increases linearly from far end to spacecraft, reaching at the spacecraft

$$T = M G(R) 3 L / 2 R$$

where M is the total mass of the tether.

The situation for dynamic stability is much more complex, since the tethered system acts as an undamped complex pendulum which moves in three dimensions according to non-rigid body dynamics. The phasing of variable disturbing forces relative to any natural resonances is critical in the essentially undamped system. Deployment and retrieval of a tethered satellite are severely constrained by conservation of angular momentum and mode coupling effects that require complex "Control Law" variations in allowable rates, particularly during retrieval. Several computer programs have been written to solve this problem for particular applications, but they are very time consuming and difficult to adapt to other situations. The problem appears to be manageable, but requires a great deal of additional attention.

One of the major objectives of the first flight of the TSS (Tethered Satellite System) scheduled for 1987 will be to validate existing tether simulation results and investigate particular aspects of dynamic behavior.

Current Coupling:

In order to complete an electrical circuit suitable for generation of electrical power, current flowing thru the wire must be carried from the upper end of the moving wire via a stationary external path to return to the wire at its lower end. The Earth's ionosphere, extending to altitudes in excess of 1,000 km, should be well suited to this function if adequate contact can be established from the ends of the moving wire to the surrounding ionospheric plasma. Some device analogous to the brushes in an ordinary DC generator is required to perform this function.

Three techniques are presently under consideration to perform this role of "plasma brush" in an electrodynamic tether power generator. The simplest is to put large conductive "halloons" at each end of the tether, to collect ambient thermal electron currents from the ionosphere at the upper end and sweep up positively charged ionospheric ions at the lower end of the tether wire. This method has no moving or electronic parts that might malfunction, however the sheer size of the conductive surfaces required becomes a significant problem, limiting its usefulness to low currents. An improvement on this method might be to substitute a large electron gun for the ion collecting balloon. This eliminates the largest of the two "balloons", but requires large amounts of power to operate and is also limited to maximum currents on the order of an ampere. Recent concepts, using plasma emitters such as the hollow cathode neutralizer systems developed for ion rockets, promise to reduce power and drag losses at both ends while increasing current capacity beyond 10A.

The "Plasma Motor/Generator":

Use of the hollow cathode plasma emitters is a crucial new concept. If such devices can be made to function as expected, they produce a much more powerful and versatile system than the original balloons and electron guns. Power consumption is reduced below 30 watts/ampere, versus 4 KW/ampere with electron guns. A 20 KW system can be designed to use a 10 km tether wire at 10A, instead of a 100 km tether at 1A. Efficiency improves to better than 90% (18KW net to load) from 50% (10KW net) for a

system with the same drag area and reduced total weight (more conductor, much less insulation and power processor electronics at 2KV than 20KV). More important is the bipolar nature of the current conduction: the same device can serve at both ends and is self-reversing as required for motor or generator operation without switching or other special control action. This allows the capability to store excess solar array power as added orbital energy during sunlight, then tapping that stored energy later to generate night-time, peak or emergency power. Ultimate growth to systems operating in excess of 100A indicates potential to handle power with >95% efficiency as either motor or generator. To distinguish such hollow cathode based concepts from the more limited balloon and electron gun design, we use the term Plasma Motor/Generator (PMG) throughout this paper.

I. ELECTRODYNAMIC TETHER POWER GENERATION

An electrodynamic tether system can be considered as a quarter-turn electrical armature in low earth orbit, cutting thru geomagnetic field lines at orbital velocity. A typical system would utilize a heavy insulated wire 10-100 km long, held radially aligned perpendicular to its orbital velocity by gravity gradient forces. The induced voltage in this wire would average 2-20 KV, depending on its length and the local horizontal component of magnetic flux.

20 KW PMG Reference System:

In order to provide a representative system to use as a reference in evaluating the performance of electrodynamic power generation in various applications, a 20 KW PMG system was designed based on use of hollow cathodes assumed to consume 500 watts while carrying a tether current of 10 amperes. The tether wire used is 10 km of #2 AWG aluminum (6.5 mm dia.) with 0.5 mm teflon insulation (based on conservative 100 volts per mil). Total mass of conductor is 908 kg, plus 99 kg insulation. The far end hollow cathode assembly is estimated at 10 kg, and 83 kg for the hollow cathode, tether controller electronics and miscellaneous hardware at the spacecraft end. Total system mass is estimated to be 1200 kg, including a 100 kg allowance for contingency. Nominal working voltage is 2,000 volts in a 400 km orbit, with up to 25% variation at various points in any given orbit. Rated power is defined as 20 KW at 10 A. Operation at higher power levels is possible, with decreasing efficiency due to increased I^2R losses in the wire until reaching a peak of 125 KW at 110 A. Tether outside diameter of 7.5 mm results in a drag area of 75 m^2 , resulting in net drag force of .045 N at an altitude where residual atmosphere density is $10\text{E}-11$ (400 km at solar max, 300 km in 1976 US Standard Atmosphere). The drag power loss is therefore $F v$, where v is the orbital velocity: $360 \text{ J/sec} = .36 \text{ KW}$. This loss must be added to the I^2R drag of 2.6 N (20 KW) produced by the 10 A current. Of the 20 KW total power produced, .50 KW is used to power the hollow cathodes, .77 KW is ohmic losses in the tether wire and ionospheric losses are estimated at .05 KW.

Performance Estimates:

The final result is 18.68 KW net power into an ideally matched load, at the expense of a total 20.36 KW mechanical power extracted from the total orbital energy of the spacecraft. Similar results can be calculated for any desired operating power. Such results are shown in Figure 1 as a plot of net electrical power to the

load versus orbit drag power for the 900 kg (wire mass) system.

The 20.36 KW results from the total operating drag of 2.63 N and will cause a loss of orbit altitude unless compensated for by equal propulsive thrust. If an H-O rocket engine of specific impulse 450 sec were used for this purpose, it would require 2.14 kg/hr to produce the 2.63 N for 18.68 KW, .12 kg/KWHR. To produce the same power using typical fuel cells is quoted at .39 kg/KWHR.

It is instructive to consider how the tether system is able to get more power from a kilogram of H-O than an extrapolated 100% efficient fuel cell (.28 kg/KWHR). The answer to this seeming contradiction is in the ability of a rocket engine to extract more than just the chemical energy of the H-O. The chemical energy is used only to heat the fuel mass in order to achieve an exhaust velocity of 4.4 km/sec. It is the resulting thrust of 2.63 N that produces work at the orbit velocity 7.74 km/sec equal to 20.36 KW. The chemical energy available for heat was only 8.4 KW. The additional energy was extracted from the orbital energy "stored" in the fuel earlier, when it was propelled into orbit. That is why a tether/rocket system can recover more than three times the power from a kilogram of H-O than a fuel cell, it is able to tap both the chemical and orbital energy of the fuel.

In principle, the chemical energy is not even necessary at all. The heat to achieve an exhaust velocity of (4.4 km/sec) could be obtained electrically using a resistojet engine running on (water or) any suitable light gas. If 8 KW were used to run such a resistojet, this would still leave 10.7 KW net power available on the spacecraft! In fact, such a system has been proposed by NASA-LeRC, to run on waste water generated by the astronauts and produce 2 KW per astronaut. It is theoretically sound, although it does require the maturing of two new technologies before it can be implemented. In any case, the tether concept with any state-of-the-art propulsion system is capable of estimated performance roughly three times that of fuel cells.

Operational Considerations:

Dynamic simulations of the reference system in operation indicate adequate margins of stability with no end mass other than the negligible 10 kg hollow cathode assembly. Working tension in the tether at the spacecraft is 21 N, at a static deflection angle of 7 degrees. Tether dynamics appear to be controllable by properly planned phasing of the operational load variations. A significant area of uncertainty exists in estimating losses associated with load impedance matching to obtain the desired power into different operational loads while working with a tether voltage that is continually changing with the local magnetic field strength and direction. This set of conditions requires the development of some sort of continuously variable DC load impedance controller. It is hoped that high frequency/high voltage devices similar to those being studied for the Electric Airplane concept will prove capable of handling this requirement with net losses less than 1%. In any case a system is required that can control tether current by matching load impedance rather than simply limiting current at either end, which would result in large power losses in the various forms of current limited sheath that would occur as a result.

The power available and efficiency of any tether generator system is mostly a function of total wire mass. Aluminum is about twice as good on a per kilogram basis as copper due to its low density. Insulation becomes a major contributor to total system mass for tether lengths greater than 10 km, becoming prohibitive for tethers much over 20 km unless special insulators are developed. Tether stability becomes marginal for tethers much less than 10 km in length, unless a very heavy gauge wire is used to also produce very low resistive losses.

100 - 1,000 KW Systems:

The 20 KW PMG performance estimates have been extended to cover a power range up to 1 Megawatt. The basic 10 km PMG remains a feasible candidate for use in Space Station applications for power levels of 100-500 KW, although the required current levels are above values typically employed with hollow cathodes (1-20 A) and are approaching values where ionospheric resistance begins to be significant. However, single hollow cathodes have been run at 50 A with no obvious problems and there is no reason to expect any substantial difficulties. For very high current applications several hollow cathodes could be placed in series along the last kilometer of tether.

At 100 A, the ionospheric loss estimates become about 5-10KW. A factor of 2 or 3 error in the resistance estimates, quite possible in fact, would make quite a difference in the higher range of efficiencies estimated using heavier tethers at this length. For this reason 20 km and even 100 km tethers were considered, particularly for use at higher altitudes where air resistance becomes unimportant.

A significant point to emphasize at these power levels is that in most cases the PMG used should be a dual system, one deployed up and another deployed down from the spacecraft. This allows single axis attitude control and/or angular momentum dumping by differential torque balancing, but more importantly it greatly reduces the total mass of a given system by reducing the amount of insulation required. A pair of 10 km PMG's operating as a dual system at 100 KW each would have the electrical performance efficiencies of a 20 km PMG, without the more than 1,000 kg weight penalty required to insulate the 20 km system's 4 KV operating potential. The power converter/load controller device is also much easier to engineer to operate at \pm 2KV rather than 4KV.

Performance estimates are shown in Figure 2, for two upsized systems using 1,800 (two of the 20 KW reference systems deployed in tandem) and 9,000 kg wire masses. The smaller system, using 1800 kg of #2 AWG aluminum wire, would provide adequate performance over the range of 60-175 KW net power provided to the load. 60KW would require an average thrust of 8.4N (65KW; 92% efficiency for conversion of mechanical power of orbital motion into electrical power delivered to the spacecraft) to maintain the orbit altitude against total operating drag. If a liquid rocket engine operating at $I_{sp} = 400$ sec were used to provide this mechanical power, a net fuel consumption of 7.7 kg/hr would be required to generate the 60 KW of power. This is .13 kg/KWHR. If net power output is increased to 175 KW, the mechanical to electrical power conversion efficiency drops to 76% due primarily to resistance losses in the relatively light wire. Orbit reboost thrust required would be 29.7 N, using 27.3 kg/hr of fuel for a net fuel requirement of .16 kg/KWHR. This is still far superior to any fuel cell system. Peak power available would be about 250 KW, at about .25 kg/KWHR.

II. ELECTRODYNAMIC TETHER THRUST for PROPULSION and POWER STORAGE

If a larger wire is used, peak power and operating efficiency increase roughly in proportion to total wire mass. At 175 KW a system using 9,000 kg of aluminum wire would operate at an efficiency of 96%, reducing the orbit reboost fuel required to 21.7 kg/hr (.12 kg/KWHR). The net saving of 134 kg/day would require 64 days to "pay back" the roughly 8600 kg greater initial system weight. However, a more immediate advantage could be gained if used for long term operation on a Space Station employing a large solar array for power. The 96% operating efficiency of the PMG could be employed in reversible operation to store part of the solar array's

output as orbital energy during daytime, then convert this back to electrical at night or to meet peak power demands in excess of 1 MW! The net storage/reconversion efficiency would be better than 91%, very competitive with battery storage systems estimated (by MSFC) to weigh more than 20,000 kg and cost over \$70M. As a fringe benefit, orbit reboost thrust for the entire Space Station/Solar Array/PMG system would be provided at a net cost of 1-5 KW (assuming total drag for the Space Station with solar array to be 0.1-0.6 N). This would eliminate the requirement for rocket engines and 1,000 to 5,000 kg/year of fuel to provide this function.

Figure 2 also shows the performance variation with altitude for two representative 20 km PMG₁₁ wire sizes. (An altitude corresponding to a residual atmosphere density of 10^{-11} kg/m³ at the lower end of the tether wire is used as the reference altitude for all drag calculations in these figures, except as otherwise noted. This corresponds to an altitude of 300 km in the 1976 United States Standard Atmosphere at an Exospheric temperature of 1,000° K, or an altitude of 400 km under the higher Exospheric temperature of 1500°K that may prevail during solar maximum conditions during the early 1990's. This is probably more conservative than most of MSFC's Space Station calculations which appear to be based on an atmospheric density of $2-3 \times 10^{-12}$ kg/m³ at orbit altitude.) Note that any of these systems would be practical for operation at extremely low orbit altitudes, with the typical efficiency "reversal" favoring the shorter, lighter systems for less than 175 KW net power at 160 km altitude due to their lower total drag area.

This suggests the interesting possibility of using a PMG to allow Space Station to be placed in a significantly lower orbit than presently planned. Adding 80 KW of net (solar array) power capacity to any proposed Space Station design would provide 10N of PMG thrust for orbit maintenance. Assuming the baseline system had a total drag of 0.1 N at 300-400 km altitude, and 50% increase in drag due to the added solar array area, the Space Station orbit could be lowered to an altitude of 160-180 km while reducing the orbit reboost fuel requirement from 800 kg/year to 350 kg/yr.

The variation with total wire mass in expected Generator performance for 100 km and 20 km tether wire lengths was calculated at power levels of 20 KW to 1MW (Figure 3). At high altitudes, very little difference is noted between 20 km and 100 km systems of the same wire mass. Total system mass, however, is much greater for the 100 km system due to the insulation and isolation requirements to operate at its 20 KV working voltage. A similar set of curves for operation as a Motor can be derived by simply substituting net power consumed for power to load, and subtracting the net aerodrag power to obtain net thrust instead of adding it to get net drag power. (For example, the 1800 kg system driven at 122 KW would produce net thrust power of $100-5 = 95$ KW, about 12 N. At 280 KW it would produce $200-5 = 195$ KW, about 25 N.)

Figure 4 shows the relative performance of existing methods of power generation (fuel cells or solar arrays) as a function of altitude; compared to projected performance of PMG based systems used either (a) purely to produce power at the expense of H₂O burned in a rocket engine rather than using the same H₂O supply to operate fuel cells or (b) to provide thrust for a solar array system in place of both rocket reboost and battery power storage. The fuel cells show no altitude dependence, since they produce no drag. They are also the lightest system in terms of initial system mass to be carried into orbit. Their fuel consumption of 390 kg/hr vs 130 kg/hr for the PMG/rocket system quickly nullifies the initial system mass disadvantage of the PMG/rocket system, even if a 1,000 kg rapid retrieval system is included for short term use. For permanent facilities, the solar array remains superior to a PMG/rocket system for altitudes above 2×10^{-11} kg/m³ (190-240 km under normal conditions) except where construction/erection time or operational attitude constraints are a factor, but it in turn is greatly surpassed by operation of the same solar array with a PMG system used to replace the batteries and orbit reboost rocket system.

Figure 5 shows the PMG system mass required to operate at a given net power and efficiency. Also shown are approximate limits due to dynamic deflection of the tether caused by the thrust/drag forces produced. For generator operation, very little mass saving is obtained by dropping below 70% efficiency, although the system can be driven to higher net thrust during Motor operation by accepting ohmic losses in excess of 50%. The question is academic for 10 km tethers, as the dynamic limit of tether deflection by the force generated is exceeded at about this same level; forcing the use of a more massive, therefore more efficient, tether wire. For the 20-100 km tethers, operation at significantly higher peak power levels is dynamically permitted. However, in most cases, a superior system could be designed by using the additional mass required to insulate the 20-100 km systems to provide increased wire mass in a 10 km system with increased efficiency and an adequate margin of dynamic stability.

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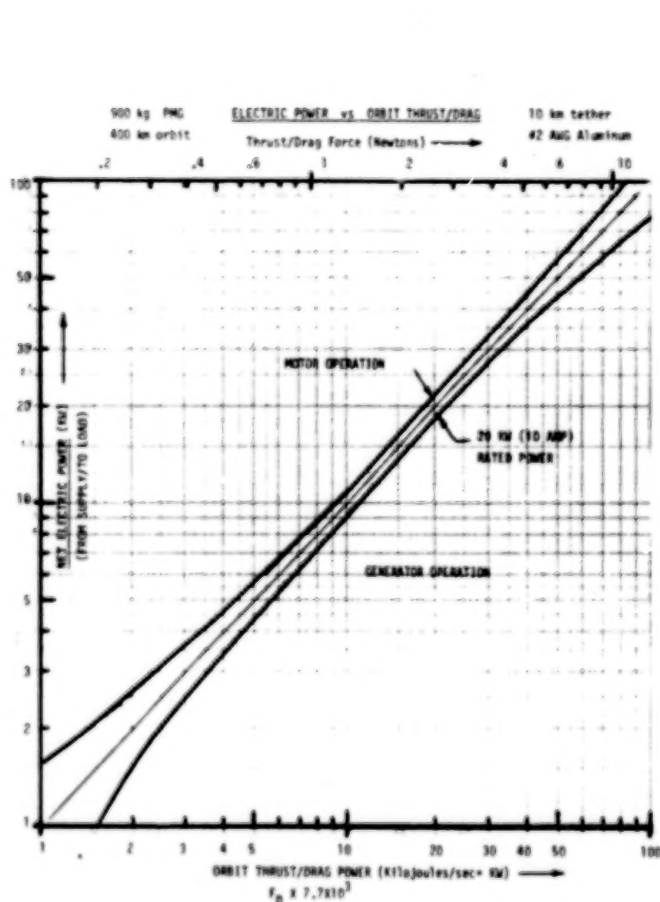


Figure 1. Plasma Motor/Generator Power

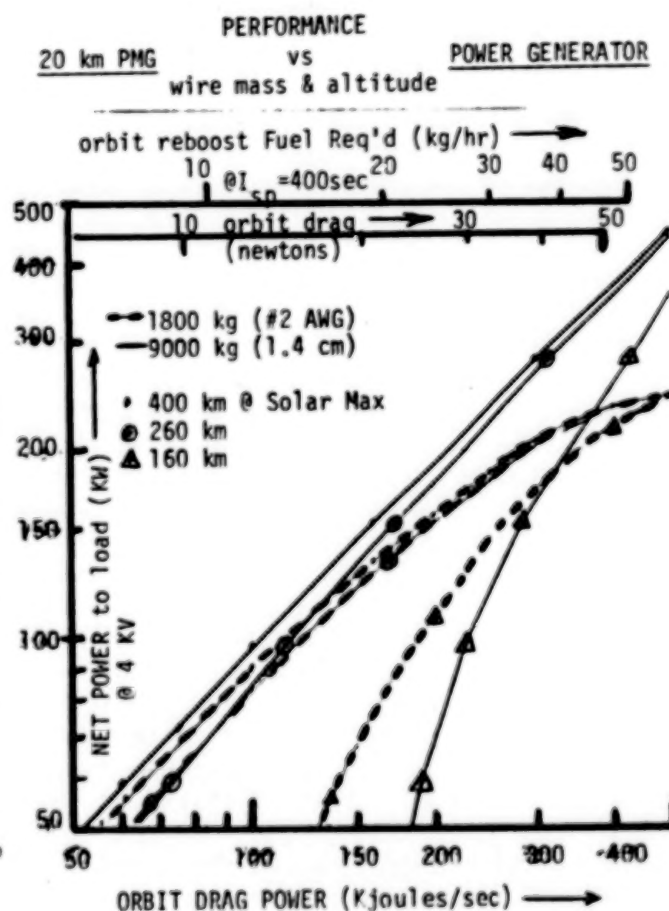


Figure 2. PMG Power Generated vs Drag

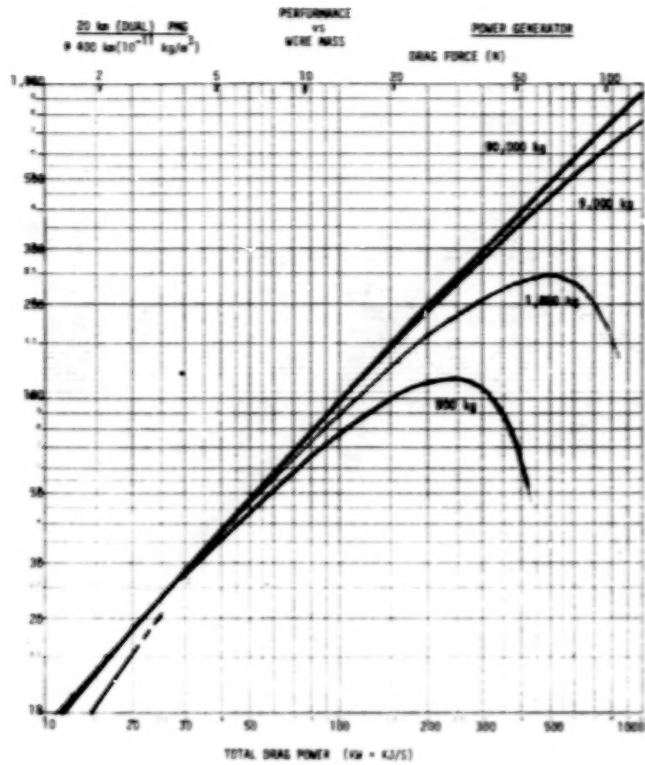


Figure 3. (90,000kg plot broken 20-50 KW)

100 KW NET POWER GENERATED
KG/HR CONSUMED vs ORBIT ALTITUDE

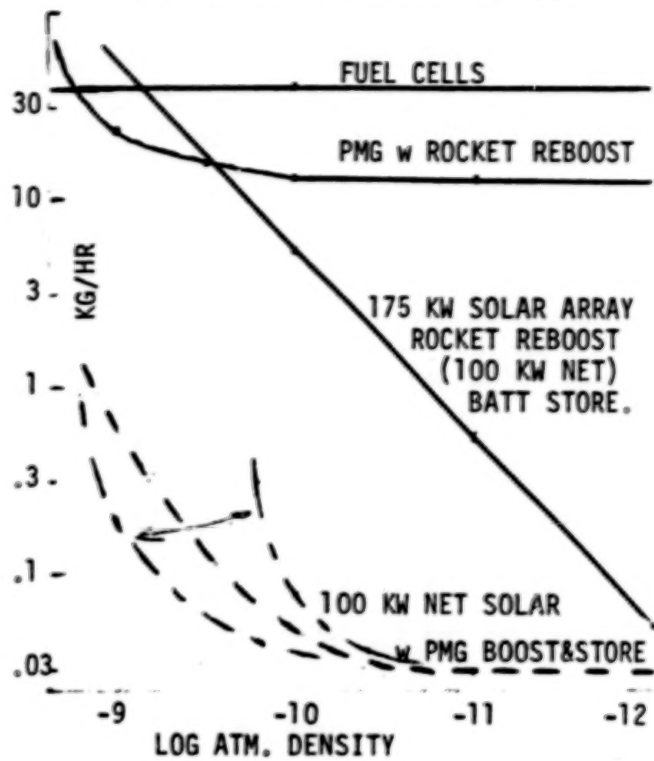


Figure 4. Mass Consumed in Orbit

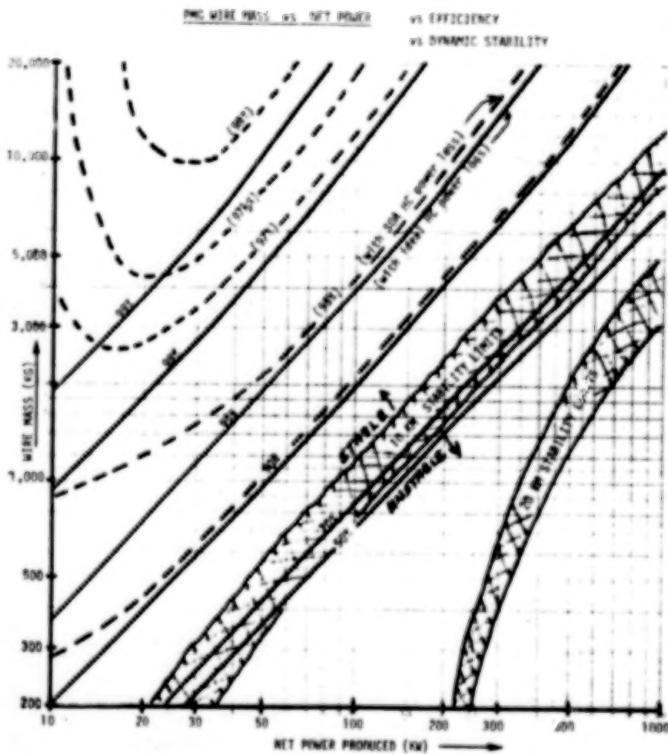


Figure 5. Performance & Stability vs Size

APPENDIX A

PMG - 20 KW REFERENCE SYSTEM

TETHER LENGTH	10 KM	WORKING TENSION	21 N
NOMINAL VOLTAGE	2 KV	WORKING ANGLE	7 DEG
RATED POWER	20 KW	RATED THRUST	2.5 N
PEAK POWER	125 KW	PEAK THRUST	>40. N
CONDUCTOR	#2 AWG ALUMINIUM WIRE DIAMETER 6.5 MM @ 20°C RESISTANCE 8.4 OHMS @ 20°C 7.7 OHMS @ 0°C 7.1 OHMS @ -20°C		908 KG
INSULATION	0.5 MM TEFLON (100 VOLTS/MIL)		99 KG
FAR END MASS	10 AMP HOLLOW CATHODE ASS'Y (INCLUDING ELECTRONICS & CONTROL)		10 KG
TETHER CONTROLLER	ELECTRONICS & MISC. HWWR. (POWER DISSIPATION LOSSES @1% = 200W)		83 KG
ARGON SUPPLY & CONTINGENCY RESERVE			<u>100 KG</u>
TOTAL			<u>1,200 KG</u>
TETHER DYNAMICS CONTROL	PASSIVE, IXB PHASING		
TETHER CURRENT/POWER CONTROL	DC IMPEDANCE MATCHING		
TETHER OUTSIDE DIAMETER	7.5 MM		
TETHER BALLISTIC DRAG AREA	75 SQ. METERS		
DRAG FORCE @ 10^{-11} KG/M ³ (300 KM 1976 USSA-400 KM SOLAR MAX)	.045 N		.36 KW
² I & R LOSSES @ 20 KW			.77 KW
HOLLOW CATHODE POWER			.50 KW
IONOSPHERIC LOSS @ 10 AMP			<u>.05 KW</u>
TOTAL PRIMARY LOSSES			1.68 KW
EFFICIENCY	ELECTRIC (18.68 KW NET @ 10 AMP/20 KW)	93.4%	
	OVERALL (20.36 MECH. TO 18.68 ELEC. KW)	91.7%	
INCLUDING CONTROLLER/POWER PROCESSOR LOSSES @ 1%			<u>.20 KW</u>
TOTAL (NET POWER OUT 18.48 KW)			1.88 KW
FINAL EFFICIENCY	ELECTRIC = 92.4%	OVERALL = 90.8%	

PHOTOVOLTAIC WORKING GROUP REPORT

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On April 10 through April 12, 1984, the NASA Lewis Research Center hosted a workshop on space power systems. This report summarizes the results of the photovoltaic sub-committee meeting where experts from government and industry attempted to answer generic and specific questions concerning their area of expertise. Twenty people attended and participated in the photovoltaic group meeting. The specific questions asked of this group are attached (see attachment #1), however, the answers are presented in narrative form here rather than in the brief presentation chart format used to present the material at the workshop.

INTRODUCTION

Solar arrays continue to be the most feasible long life satellite power source and are projected to satisfy even the multi-hundred kilowatt mission needs of the 1990's. Development and operation of several hundred solar array/battery powered satellites has advanced solar cell and solar array technology significantly. Even so, today's specific power range of 10-35 watts per kilogram (W/kg) for solar arrays is projected to increase to 180-240 W/kg by 1995. Thin silicon, thin film gallium arsenide (GaAs), multijunction, and concentrator optimized solar cells, combined with flexible or concentrator solar array substrates are areas identified where work must be done to meet the 180-240 W/kg goals.

Other very significant goals such as longer life in orbit, survivability from both nuclear and laser threats, autonomy, and resistance to degradation from interaction with the space plasma environment will require a significant amount of additional development work.

If the 1995 specific power goals can be met, photovoltaics will continue to provide the power needs for the vast majority of satellite missions in that time-frame.

ANSWERS TO GENERIC QUESTIONS

The space power workshop attendees presented information and general data on projected 1995 missions in the three major sectors, i.e. military, public, and commercial. The P-V committee determined that the needed characteristics of photo-

voltaic array technology to enable these projected missions as follows:

Military

- (1) Survivability to laser and nuclear threats
- (2) Long life (up to 10 years) reliability
- (3) Power levels up to 60 kilowatts for space based radar
- (4) Radiation resistant cells for use in mid altitude orbits

Public

- (1) Sizes up to 400 kilowatts
- (2) Long life reliability
- (3) Cost effective designs
- (4) Compatibility with plasma environments
- (5) Operational toughness

Commercial

- GEO -- (1) High performance
- (2) Long (up to 20 years) life
- (3) No environmental interaction problem
- LEO (1) Low recurring cost
- (2) Maintainability
- (3) No environmental interaction problems

From these needs, it was decided that the major technology and mission constraints were:

- (1) Lifetime in mid altitude orbits -- Space Based Radar (SBR)
- (2) Higher power levels in GEO -- Direct Broadcast Communication Platform
- (3) Robust arrays for comet and asteroid missions
- (4) Higher power levels in LEO -- Space Station

Considering all of the major mission drivers, high power levels will dictate operation at high voltage and also operation in adverse environmental orbital conditions. Critical barriers to using photovoltaic systems in these environments include:

- o Effects of radiation damage
- o Arcing/leakage at high voltage
- o Atomic oxygen erosion
- o Atmospheric drag
- o Thermal cycling
- o Corona effects
- o Contamination from attitude control waste dumps
- o Combined effects from all of the above.

Studies in some form are underway on all of the above problems and flight experiments such as SAFE (Solar Array Flight Experiment), Long Duration Experimental Flight (LDEF), and VOLT-2 will answer many of the environmental effects problems. SAFE will demonstrate advanced, lightweight solar array technology; validate dynamic, thermal, and electrical models and design techniques, and answer many of the questions concerning atomic oxygen effects on Kapton. LDEF has samples of many advanced solar cells and solar array assembly sections. It also features an experiment capable of exposing cell assemblies to high voltages and measuring operational effects. Goals of the VOLT-2 experiment which is a refurbishment and reflight of the SAFE (Solar Array Flight Experiment) are: (1) Evaluate the impact of interactions on performance of large planar solar array with self generated voltages; (2) Determine floating potentials for true distributed voltage array; (3) Measure power loss and arcing threshold and impact on performance as a function of solar array voltage; and (4) Validate phenomenological and system level models which predict array performance.

The approaches needed to provide the enabling technology for using photovoltaics on 1995 missions will include concentrator arrays to enhance survivability, sizing up of solar cell and structure technology to be ready for multi-100 kilowatt missions, and addressing the environmental interaction problems associated with higher operating voltages.

The major modification to existing "technology program" directions should be to continue development work on silicon solar cells and arrays as well as doing the extensive work planned for GaAs and other advanced concept cells and arrays.

ANSWERS TO PHOTOVOLTAIC QUESTIONS

Concentrator arrays promise reduced solar array area because of both higher beginning of life efficiency and end of life efficiency. The inherent shielding of the cells by the concentrator elements also promise increased survivability from weapon effects and exposure to the high radiation environment associated with "belt flyers". Development of small array modules is currently underway in both Air Force and NASA technology programs and performance and survivability testing is planned and has been conducted on the Cassegrainian and SLATS concepts. The further developments required to demonstrate concentrator capability in space include fabrication of high precision array structures, development of stowage and deployment techniques, and precise pointing ability to $\pm 1^\circ$ for the large array areas. This hardware must be ground tested and preferably flight tested also on experimental vehicles. Present concentrator elements are costly to produce, and it is strongly urged that less expensive fabrication techniques be developed and their suitability demonstrated. For low earth orbit severe environmental interactions are expected from atomic oxygen, the space plasma and contamination from various wastes and outgassing of the space station or other large satellites. Also since concentrator elements are largely bare metal, currents are expected to flow between these elements and the plasma causing power loss, electrical transients, and possible power disruption. This condition is likely to be more severe at the higher voltages needed for the higher power system requirements. The heart of the concentrator systems is the solar cell. Performance and contact integrity need to be demonstrated in ground tests and long life flight

experiments. The surfaces of reflectors need also be evaluated. Tests should be both real time and accelerated. While this concentrator technology is relatively new it shows high promise for life, cost, performance, and survivability factors and is well worth the resources needed to demonstrate its viability.

It is anticipated that ultrathin silicon solar cells with gridded back contacts can be developed to deliver 15 percent conversion efficiency at operating temperature in space. Assuming an array packing factor of 0.9, this corresponds to an area specific power of 100 W/m^2 . By employing welded interconnects, ultrathin covers ($\leq 50 \mu\text{m}$), flexible substrates and spot bonded adhesive techniques, array blankets with a specific weight of 0.5 kg/m^2 can be achieved. Combining this blanket with a lightweight structure such as the Astromast or STACBEAM (stacking triangular articulated compact beam) which will, for high power arrays ($> 20 \text{ kw}$), have a weight equal to that of the blanket, (0.5 kg/m^2) will yield an array specific power, at beginning of life, of $\sim 180 \text{ W/kg}$.

Further improvements in array specific power can be realized if ultrathin GaAs cells can be developed. It has already been demonstrated that $5 \mu\text{m}$ GaAs cells are capable of providing the same conversion efficiency as thicker ($> 300 \mu\text{m}$) devices. Greater than 18 percent conversion efficiency has already been demonstrated for GaAs and it is likely that 20 percent can be achieved within the next 3 to 5 years. By utilizing the same approach to lowering the cell's operating temperature as has already been demonstrated for silicon, a 16 percent efficiency at operating temperature is not an unrealistic forecast. This yields a real specific power of $\sim 240 \text{ W/m}^2$, which when combined with lightweight blanket and structure advancements, provides an array with a beginning of life specific power of $\sim 240 \text{ W/kg}$.

The technological approaches which are needed to fabricate and operate solar cell arrays (planar, flexible or concentrator) at sizes beyond 200 kw involve many disciplines. The obvious need is for deployment and support structures and the system to orient them. Also the arrays will need to operate at voltages in the 200 to 400 volt range. Power transfer techniques for high currents and voltages across the interface between the array and its associated vehicle and mission loads are needed. Concepts which may be investigated include rotary transformers, sealed slip ring assemblies or ring transfer concepts. Since large areas of cells are needed and development of a single system is expensive we need modularity in system assembly to match the load requirements and allow for easy expansion. Larger cells and automated assembly techniques are also needed to keep the costs within reasonable levels.

GaAs solar cell radiation damage behavior needs to be determined on the kinds of cells that will be used on a given mission. This means a quantity of 500 or more uniform statistically significant solar cells of each production type need to be tested. Efforts are underway on the LPE fabricated cells from Hughes Research Laboratory to determine the effects of a wide energy range of proton and electron particles both omnidirectional and normal incidence. This will establish generic GaAs cell behavior and establish damage equivalence (modeling cell energies equivalent to 1 MeV electrons) and determine if the technique is viable for GaAs solar cells. This

data needs to be widely published in a handbook or similar form. The space environment itself needs more precise modeling. This is especially true of the lower Van Allen belt and the solar flare environment. The environment from low earth orbit to near synchronous will be extensively mapped by the Chemical Release Radiation Effects Satellite (CRRES) when it enters its second flight orbit 400 X 35, 800 km elliptical, 23° inclination. A GaAs flight experiment is included on this vehicle to measure degradation and to determine if real time annealing is possible. Three annealing modes are included. The solar flare environment can produce a major part of the radiation in geosynchronous orbit. We need a more accurate method of predicting solar flares. Perhaps the problem is a scarcity of information on the physics of solar flares. Solar physicists should be stimulated to achieve a realistic model of interactions and mechanisms within the sun which cause solar flares. What can "Solar Max" tell us? In addition to the degradation of the cells we also need information on the rest of the array materials on exposure to the radiation environments.

Reasonable efficiency and cost goals for GaAs solar cells are summarized in Table 1 below. Concentrator cells should be considerably cheaper for equivalent output because of their smaller size and higher power density.

Table 1 - GaAs Efficiency Goals

<u>CONCENTRATOR</u>				
<u>NEAR TERM</u>	20%	GaAs	@ 100X	80°C
<u>INT TERM</u>	22%	GaAs	@ 100X	80°C
	21%	GaAs	@ 20X	80°C
<u>FAR TERM</u>	30%	CASCADE	@ 100X	80°C
	25%	SPECTRUM SPLITTING	@ 100X	80°C
<u>PLANAR</u>				
<u>NEAR TERM</u>	17%	@ 25°C		
		\$100/2 x 4 CELL		
<u>INTERMEDIATE TERM</u>	20%	@ 25°C		

COST IS SUBSTRATE AND YIELD DEPENDENT

THIN CELL + GaAs = SI COST IN FAR TERM

The basis for future R&T work are myriad. The next, 17th, Photovoltaic Specialist Conference includes many new concepts and experimental programs that show promise for potential space use. These included heterojunction silicon and GaAs based cells. Thin and thin film GaAs cell work should be pursued which could lead to dramatic reductions in solar cell weight and improved performance. The amorphous

silicon cells and associated tandem concepts are reporting AM 1-2 efficiencies of 18-20%. With further understanding of these cell mechanisms even better performance may be achievable. Also the II-VI areas are showing some progress after 30 years of research. In particular the CuInSe₂ type of cells show good stability and the ability to readily achieve low loss stable tunnel junctions for tandem or cascade cell combinations. The last area identified, not necessarily the only one, is to improve solar array end of life performance through better understanding of cell degradation mechanisms and to pursue technical approaches to minimize these effects.

ENVIRONMENTAL INTERACTIONS

Radiation Damage This is a long recognized phenomena and is addressed elsewhere.

Arcing and Leakage Measurements in the lab and in space have shown there is significant current flow between bare metallic conductors and the plasma environment in low orbit. This increases almost exponentially at voltages above 100 volts and is more severe when the array is positive with respect to the vehicle. The arcing - especially at higher altitudes, can be caused by discharge of electrons trapped in dielectric surfaces such as coverglasses, insulation or dielectric radiator/thermal coatings.

Atomic Oxygen Erosion This has been observed from the shuttle on many organic and plastic materials such as mylar, kevlar, and epoxies in the matrix materials.

Atmospheric Drag and Thermal Cycling Well documented

Corona Effects The environment around and within a spacecraft is not always a high vacuum because of material outgassing and the gravitational or electrostatic "halo" effect. Thus, as we operate with higher voltages on arrays and within power systems corona can easily occur with the resultant EMI noise and power losses. This is especially true for the conventional slip ring array power transfer technique and needs to be addressed early on.

Contamination There are several sources of materials which are ejected into the environment around a spacecraft and the solar array such as adhesive outgassing attitude control and propulsion jets and waste dumps. These effluences can lodge on solar array surfaces and cause many effects including obscuration - reduced light transmission, change absorption or emission properties and lead to electrical breakdown. Chemical breakdown may also occur if these materials become ionized by the plasma or solar UV environment.

Combined Effects Are simply the synergistic effects of all the above.

Attachment #1

A. Generic Questions

1. What are the needed characteristics of your technology to enable the missions in the three sectors (military, public, commercial) in the period 1995 - 2005?
2. What missions are constrained by the existing level of the technology?
3. What are the technology constraints for the missions identified in question 2?
4. What mission(s) is(are) the strongest driver(s) for the technology?
5. What are the current significant technology deficiencies and critical barriers in your technology area?
6. Where are new technology approaches needed? Will these approaches provide enabling or enhancing technology?
7. What is the adequacy of current and planned programs?
8. What additional tasks should be done? What are the benefits? What is the necessary timing?
9. What are the lowest priority items in the present and planned programs?
10. What potential problems do you foresee regarding the interaction between your hardware and the environment that may be encountered in use. This can include contamination by mission operations as well as the natural environment.

B. Photovoltaic Questions

1. What technological developments are required to make concentrator arrays more viable in space?
2. What are reasonable specific power goals for the '90's for silicon and gallium arsenide arrays?
3. What critical technology advances are required to reach these goals?
4. What are the technological approaches leading to array sizes beyond 200 kw?
5. What needs to be done to allow one to predict the radiation damage to gallium arsenide solar cell arrays for a given mission?
6. For what orbits is radiation damage to solar arrays critically important? What additional information, if any, is needed for predictions of array performance in these critical orbits?
7. What are reasonable efficiency goals for concentrator solar cells?
8. What are reasonable efficiency and cost goals for gallium arsenide planar cells?
9. Do any advancements in fundamental understanding provide a basis for new conversion device R&T work?

ELECTROCHEMISTRY AND STORAGE PANEL REPORT

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In the early 1960's, both NASA and the Department of Defense (DOD) recognized the potential gains to be made in spacecraft weight, volume, and endurance characteristics by the development of improved electrochemical energy storage systems. The first of these systems to reach flight qualification status was the individual pressure vessel (IPV) nickel hydrogen cell, which is now baselined by commercial users as well as by the government for geosynchronous orbit applications.

Today, as power levels orders of a magnitude higher than current spacecraft requirements are contemplated by NASA and DOD, it has become obvious that another generation of improved energy storage devices is required. Electrochemical couples and systems with the potential of meeting these requirements have been identified, and exploratory work has begun. Examples include the sodium-sulfur (Na-S) battery currently under investigation by the Air Force Aeropropulsion Laboratory and the regenerative hydrogen-oxygen (H_2 - O_2) system currently under development by the NASA, Lewis Research, and Johnson Space Centers.

In addition to the development of these higher energy systems, an improvement in the performance and durability of the nickel hydrogen cell is expected to be achieved through a better understanding of the fundamental electrode mechanisms. It is predicted that the volumetric and gravimetric energy density of electrochemical energy storage systems will improve quantitatively by 200 to 400 percent in the next 15 to 20 years. This improvement in performance will allow the government and commercial sectors to meet future mission requirements with space-proven photovoltaic/electrochemical energy storage power systems rather than be forced to higher risk and unproven power sources.

This report attempts to provide general answers to the questions asked by the Workshop Chairman. The limited time available to the panel required that the questions be grouped into areas of common issue.

TECHNOLOGY CONSTRAINTS AND APPROACHES

Geosynchronous Orbit - Space Shuttle/Centaur-G geosynchronous launch capability is approximately 10,000 pounds. With present nickel hydrogen IPV cells for energy storage apportioning the power system weight to that of a typical satellite, the power level would be limited to under 15kW. Several projected DOD and NASA missions require over 25kW. Therefore, an improved energy storage system is an enabling technology for future geosynchronous orbit missions.

A goal for the NASA Technology Program is to increase usable battery energy density in GEO by a factor of two or more while achieving a highly reliable ten-year life. It is estimated that the energy density of present-day Ni/Cd and Ni/H₂ batteries are 20 and 30 Wh/Kg, respectively. A reasonable goal for an advanced battery system is 60 Wh/Kg. The recommended approaches to the development of an advanced battery system for geosynchronous orbit include development activity toward advanced regenerative H₂-O₂ fuel cell system and high temperature batteries such as Na-S and Li/FeS metalsulfide. Research activity on solid oxide and polymer/lithium secondary cells is also recommended. Because an evaluation of the potential fly-wheels for this application is not found in the literature, it becomes necessary to assess their full potential.

Low Earth Orbit - The high cycle requirement (5000 per year) required in low orbit severely limits the allowable depth of discharge of present batteries in order to achieve the desired 3 to 7-year endurance. Improvements are required to reduce weight (increase depth of discharge) on future missions. These future NASA and DOD missions will require that power levels be ordered in quantities higher than those of present day LEO spacecraft. This will require the development of larger and higher voltage energy storage systems. For the battery systems the goal is to increase the energy density (by increasing allowable depth of discharge) by a factor of 2 to 4 for a 7-year-life (30,000 cycles). For regenerative fuel cells, the goal is an increase of 10% in the charge/discharge efficiency. Approaches to these goals include:

- (1) Fundamental studies of the electrochemical and degradation phenomenon of the nickel electrode should increase the allowable depth of discharge to 50%, thus improving the performance of the Ni-H₂ and Ni-Cd systems.
- (2) The regenerative fuel cell life and efficiency will be increased by development of improved catalyst and materials research to permit higher temperature operation.
- (3) Longer range, it is recommended to support exploratory work on H₂ - Halogen systems, solid oxide H₂-O₂ systems, and flywheels.

Radar Type Loads - These loads are characterized by high peak to average/low profiles and high frequency pulse-loading of the power systems requiring fast rise times from the battery. Because of internal impedance considerations, present day batteries require oversizing to meet these requirements. It is recommended that for these types of loads bi-polar systems be developed, since they have inherently lower internal impedance and are more easily adapted to high voltage system requirements.

It is significant that the fuel cell is currently a bi-polar concept because bi-polar designs for nickel-hydrogen and lead acid batteries have been proposed and are receiving development attention.

High Voltage Systems - Future high power satellites will require energy storage systems with output voltages of over 100V and possibly much higher. It was the panel's consensus that bi-polar batteries and fuel cells are most amenable to meeting this requirement because the cell interconnection is made internally to the battery rather than via external wiring. Multi-cell series connection of individual cells into high voltage batteries is a viable approach but will result in a weight and volume penalty to the spacecraft.

Flywheel Technology - Recent studies of flywheels for low earth-orbit application show the flywheel to be a potentially viable alternative for high voltage system requirements. Open issues include the possible beneficial effect of integration of flywheel energy storage with the attitude control system and the safety concerns dealing with containment and torquing imbalance.

Primary Batteries - Several applications such as get-away-specials, space suit and life support, and portable tools require small lithium cells. Development of safe Li/SOCl_2 with a performance of 300 Wh/Kg at rates of over C/5 are required for future space missions. The recommended approach to this development is to continue work toward a fundamental understanding of the chemical processes involved and to develop a new cell design; then, to transfer this technology base to the manufacturing community.

Higher power primary cells and batteries are required for high power probe and emergency power applications. Goals in this area include a high reliability, sterilizable multi kilowatt "reserve" system and high energy/high power primary batteries. The recommended approach in this area is to evaluate hydrogen/oxygen fuel cell technology for the high power primary battery requirement and to evaluate high rate Li/SOCl_2 and $\text{Li/SO}_2 \text{ Cl}_2$ reserve batteries for the sterilizable requirements.

Programmatic Issues - Three programmatic issues were discussed by the panel: Significant lead time is required to verify the endurance potential of a new electrochemical technology prior to application in a flight program. In addition, flight verification of the new concept is required as part of the qualification procedure. These factors imply long and costly programs to implement a new technology into standard use. The second issue is the need of a mechanism to motivate program offices to upgrade the technology as it becomes available from other development activities. Historically, it has been very difficult to incorporate new technology when it only enhances the mission rather than enables it. The third issue was the difficulty in maintaining reliability and pedigree of flight-qualified hardware if manufacturing process or even possibly vendor base changes occur over the course of a program. No clear answers were presented for these issues. They are, therefore, presented in this report as unresolved questions for future thoughts.

NASA Program Plan - Based on its limited visibility into the planned NASA program to 1995, the committee saw no major discrepancies in the technology areas covered by the plan and the technologies areas discussed at the Workshop. For example, the plan addresses work in new areas such as rechargeable lithium batteries and also contains supporting technology work for the nickel hydrogen and regenerative fuel cell systems, both of which were recommended by the panel.

DYNAMIC CONVERSION WORKING GROUP SUMMARY

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This Working Group addressed the potential of dynamic conversion devices for use in solar and nuclear dynamic space power systems. Conversion systems considered were based on the use of Brayton, Stirling and Rankine cycles. Both organic and liquid metal Rankine cycles were included.

The basic dynamic conversion system considerations addressed by the working group were: mission requirements, system attributes, system options, technology issues and constraints, and priorities of needed technology development. Mission requirements, where dynamic conversion was considered enabling technology, were identified along with the associated power levels and potential energy sources. When considering the system options special attention was given to recommend operating temperatures and other significant discriminators. The final result of the working group was a list of prioritized tasks considered important for the successful development of dynamic conversion systems for 1995 and beyond.

Potential Missions

In general it was felt that applications that would require 100 KWe or more would be enabled by the dynamic systems technology. These missions would include the growth Space Station, GEO Communications Platform, GEO and Lunar Payload Delivery, Interplanetary Travel, Far Outer Planet Orbiter, Multi-Asteroid Sample Return, and Lunar and Asteroid Resource Utilization. There are also a number of military missions which would require power levels high enough to require the use of dynamic systems. The heat source for all these missions was considered to be nuclear with the exception of the growth Space Station which was considered to be solar with a nuclear option. It was felt that there were no technology issues which precluded application of solar dynamic systems to the baseline Space Station. It was recognized that many engineering design issues exist, and that in some cases the line between engineering design issues and technology issues was debatable.

For some low power missions, such as communications or planetary exploration, where power levels of 10-20 KWe are required, the use of dynamic systems with an RTG heat source were considered applicable but not enabling. It should be pointed out, however, that the members of the Working Group generally considered the

dynamic systems as enabling a mission when compared to solar PV systems. The use of nuclear thermoelectric and thermionic systems are also being investigated to meet many of these missions.

Attributes of Dynamic Systems

Attributes of dynamic conversion systems that make them enabling or highly attractive relative to the mission application considerations included the following. The power level growth potential to the MW range was a basic enabling attribute. The state of the technology as proven by the excellent low temperature ($\sim 900^{\circ}\text{K}$ - 1100°K) data (experience) base was another key attribute. Also, there is a great potential for integrating the dynamic conversion technology with other technologies, particularly in the areas of thermal energy sources and storage. Other positive characteristics of these systems include their efficiency, suitability for "hardening" for military applications, direct generation of AC power, and for nuclear source systems an independence of solar flux.

Dynamic Systems Considered

Viable dynamic system options were defined as inert gas Brayton, organic and liquid metal Rankine, and Helium Stirling thermal cycle machines operating with solar heat sources at temperatures below 1100°K . Also, Brayton systems operating in the 950 - 1500°K range, Stirling systems in the 900 - 1500°K , and liquid metal Rankine systems in the 1100 - 1350°K range were considered viable options using nuclear heat sources. Within these options, "first system" peak temperatures of 1100°K for the solar dynamic Brayton systems, 670°K for the organic Rankine systems, and 900 - 1100°K for the nuclear Stirling or Brayton systems were recommended. Between the solar and nuclear options, it was felt that the solar systems would be lighter for manned applications and the nuclear systems lighter for unmanned applications. This was based on consideration of the initial mass to orbit for these systems. One must also consider the propellant resupply necessary for drag cancellation over the lifetime of the missions for the manned systems operating in LEO before choosing between the nuclear and solar dynamic systems.

Technology Issues

Technology issues, constraints, and engineering development issues for the dynamic conversion systems were defined for both solar and nuclear heat sources.

In general it was felt that there is an excellent data base available for Brayton Systems operating at temperatures below 1100°K and that there are no technology issues for the power conversion systems at these temperatures. Similarly, Organic Rankine Systems have been operated (terrestrial) at temperatures of $\sim 670^{\circ}\text{K}$. Also, essentially all of the components for liquid metal Rankine systems operating at temperatures of 1350°K or lower have been tested for thousands of hours.

The technological issues identified revolved around the development of a suitable heat source, solar or nuclear, and operation at temperatures above 1100°K . The one exception was the Stirling Engine which is a somewhat less mature technology than the other systems. Considering the heat sources, development of lightweight, deployable, erectable solar concentrators, mirror degradation, thermal receiver

technology, and thermal energy storage were concerns expressed for the solar dynamic systems.

For the nuclear systems compact, long-lived reactors for space power systems must be developed. This task gets increasingly difficult as the reactor outlet temperatures are increased from $\sim 900\text{K}$ to 1550°K .

For the organic and liquid metal Rankine systems a major feasibility issue was the operation of two-phase flow systems in a zero gravity environment. The Brayton cycle issues were found to be a function of the operating temperature. Below 1100K no issues were identified, at $1100\text{--}1500\text{K}$ refractory metals are required and concerns expressed included materials characterization, and inert gas compatibility/lifetime characteristics. Above 1500K development of ceramic components will be required.

For the Stirling Cycle, larger (25 KWe) free piston engines must be developed, reasonable efficiencies must be demonstrated at a temperature ratio of two or less and hydrodynamic or hydrostatic bearing feasibility must be demonstrated.

It was felt that all systems would benefit from the development of deployable/erectable radiators and/or advanced radiator technology.

Technology Development Recommendation

In conclusion, the following engineering and technology development efforts were recommended. They are listed in order of their priority.

- a. A near-term flight experiment using one of the more developed systems (solar Brayton or organic Rankine) to establish some space operation data base. It was felt that there are a lot of "perceived" problems with dynamic conversion systems that are not real and that a flight experiment would do much to dispel these fears and establish dynamics as a viable space power system option.
- b. Development of solar heat source technology including collectors/concentrators, receivers, thermal storage/extraction, and thermal energy transmission (thermal bus) systems.
- c. Development of nuclear heat source technology.
- d. Stirling system state-of-the-art technology development.
- e. Development of advanced radiator technology.
- f. System integration studies.
- g. High temperature conversion system technology development including materials, radiators and heat pipes, and thermal storage.

THERMOELECTRIC AND THERMIONIC CONVERSION TECHNOLOGY

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NASA Headquarters

A thermoelectric and thermionic working group consisting of NASA and industry personnel was convened at NASA Lewis Research Center on April 10, 11 and 12. The working group members heard presentations by NASA, DOD and commercial companies on the potential space missions beyond 1995 and were asked to answer a set of questions relating to the future technology needs of thermoelectric and thermionics to support the potential missions. The working group reviewed the missions, reviewed the present state of thermoelectric and thermionic technology and recommended applied research and technology efforts be pursued for thermoelectrics, thermionics and power electronics technologies to support the potential missions beyond 1995. The working group recommends the effort in thermoelectric materials technology be broadened with the objective of obtaining a material with a Figure of Merit greater than $1.0 \times 10^{-3}/K$ for hot junction temperatures of 1100 to 1500 K. The recommended effort in thermionics is to pursue technology programs that will result in an understanding and lifetime prediction methodologies for fuel-emitter and sheath-insulator behavior as a function of operating time and temperature. Also the working group recommends an effort be initiated that combines the thermoelectric, thermionic and power electronic technologies into a program to develop the technology for high temperature, high radiation resistant, high current electronic switches.

INTRODUCTION

The purpose of this paper is to recommend applied research and technology efforts in thermoelectric and thermionic conversion programs that will enable space power systems for the nation's future space missions beyond 1995. The potential space missions could be for NASA, commercial and/or military applications. The heat source could be solar or nuclear. An overview of the potential space missions beyond 1995 were presented at the workshop and are included in these proceedings. The thermoelectric and thermionic working group consisted of the authors of this paper as co-chairmen and the following technical experts;

Richard Dahlberg, GA Technologies

Richard Katucki, GE Missile & Space division

Ira Myer, NASA Lewis Research Center

Charles Wood, Jet Propulsion Laboratory

Don Chubb, NASA Lewis Research Center

After hearing papers on the technology status of thermoelectrics and thermionics, the working group was given two sets of questions, one generic for all working groups (see Table 1) and one specific to thermoelectrics and thermionics (see Table 2). This paper summarizes the working groups answers to those questions and a recommended applied research and technology approach for future thermionics, thermoelectrics and power electronic programs.

THERMOELECTRIC CONVERSION DISCUSSION

The military, public and commercial missions in the time period 1995 to 2005 appear to be at power levels of 50 to 500 kWe and require specific power of 25 to 50 watts/kg for unmanned missions, with the second generation space station being the only manned applications in the 1995 to 2005 time period and requiring 10 to 20 watts/kg at a 300 kWe power level. A conceptual design of a thermoelectric converter that would be useful for these future missions is illustrated in Figure 1. The key technology issues of materials, couples and converters for thermoelectrics are also designated in Figure 1. The needed characteristic of thermoelectric (TE) conversion to enable these potential missions is a Figure of Merit ($FOM = S^2/Pk$, S = Seebeck voltage, P = electrical resistivity and k = thermal conductivity) greater than $1.0 \times 10^{-3}/K$ with hot junction temperatures of 1100 to 1500 K, and operating lifetimes of 10 years. The missions constrained by the existing level of technology are those requiring greater than 100 kWe and specific power greater than 25 watts/kg. Thus the thermoelectric FOM and converter specific power needs to be improved to enable the second generation space station at 300 kWe, the electric propulsion missions requiring 40 watts/kg and the military missions requiring specific power greater than 25 watts/kg..

The missions that appear to be the strongest drivers for the TE technology improvement are in priority:

- A) Military Surveillance mission, i.e., 100 kWe space radar
- B) Military Communication mission, i.e., 100 kWe laser
- C) Second Generation Space Station mission i.e., 300 kWe, material processing, and
- D) Electric Propulsion missions i.e., NEP Tug and Outer Planet Exploration

The current critical barriers in the thermoelectric technology are materials with a FOM greater than $1.0 \times 10^{-3}/K$ operating at hot junction temperature of 1100 to 1500 K with temperature drops of 500 K.

THERMOELECTRIC TECHNOLOGY APPROACH

The new technology approaches needed are a broader TE material effort at 1100 to 1500 K, emphasizing lower temperature because of the reactor heat source technology limits and new material fabrication processes applied to Si-Ge, La-S, B-C and similar semiconductor materials to obtain very small stable grain size less than 1 micron (approx. 400 angstroms) to reduce thermal conductivity. Also more University involvement is recommended in exploring new ideas for better TE materials. The current and planned technology program for TE is adequate to support the SP-100 Project. However, the applied research and technology effort needs additional support in order to have a broader base technology program on TE materials at 1100 to 1500 K hot junction temperatures and 600 to 1000 K cold junction temperatures.

Also the TE applied research and technology effort needs support in designing and developing a low temperature drop coupling between heat source and TE hot junction. There is a need to support analytical and experimental investigation of TE converter designed for low cost, high reliability and mass production that can be integrated with solar, reactor and/or radioisotope heat sources. There are no low priority tasks in the present and planned TE program. The potential problem for TE in the space environment is the effects of spacecraft contamination on voltage across insulators, and on high emissivity thermal radiation coatings.

The threshold for success for TE to enable the potential mission described above are $FOM = 1.0 \times 10^{-3}/K$ with hot junction temperature of 1300 K. Si-Ge, La-S and B-C are TE materials that might achieve these thresholds. The maximum reasonable power level to which thermoelectrics can be applied is about 500 kWe. The power processing is not a critical problem for large TE systems. However high temperature (500 to 800 K), high radiation resistant (10^7 gamma and 10^{13} nvt) and high current (500 to 1000 amps) devices could significantly improve the specific power for large (100 to 500 kWe) TE systems. An applied research and technology program for power electronics should investigate using Si-C and B-C materials for high temperature, high radiation and high current components.

THERMIONIC CONVERSION DISCUSSION

A conceptual design of a thermionic converter for an in-core reactor space power system is shown in Figure 2. The key technology issues are also designated in Figure 2. The needed characteristics of the thermionic technology to enable the military, public and commercial missions prioritized above are longer lifetime, higher voltage sheath insulators, and improved performance at emitter temperatures lower than 1600 K. The existing technology constrains the missions to about 2 year lifetimes. The technology constraints which limit lifetime are fuel-emitter growth, metal-ceramic seal cracking and sheath insulator electrical resistivity degradation.

The missions that are the strongest drivers for thermionic are the same missions as the thermoelectrics plus higher power missions (megawatt class i.e., lunar base (2/MWe), manned electric propulsion (5/MWe), and potential for 10MWe military missions.

The current critical barriers for the thermionics technology are understanding the nuclear fuel-emitter distortions at 1600 to 2000 K and understanding the electrical insulator behavior at 800 to 1200 K, when bonded between niobium metal sheaths, with high voltage gradient and operating in a high neutron and gamma radiation environment.

THERMIONIC TECHNOLOGY APPROACH

The new technology approach for these two critical barriers are to develop fundamental understanding and an analytical prediction methodology for fuel-emitter and sheath-insulator behavior as a function of operating time and temperature. Also develop the technology for smaller emitter diameters (approx. 0.6 to 0.8 cm) and smaller emitter to collector gaps (~ 0.25 cm) for 1 to 5 MWe power systems. Once the fuel-emitter behavior is predictable for 1600 to 1800 K then an understanding of the fuel-emitter behavior from 1800 to 2100 K is desired. The basic behavior of nuclear fuels (UO_2 and UN) should be developed with an applied research effort extending the doctoral thesis work of Zimmerman. A parallel effort would be to investigate and develop stronger emitters, such as silicon carbide vapor deposited liners inside of tungsten emitters, tungsten-rhenium wire reinforced tungsten vapor deposited emitters, etc.

Once the sheath-insulator behavior is understood at 800 to 1200 K and at 1000 volts/cm, then insulators for higher voltage gradients and possibly higher temperatures should be investigated for higher power systems operating at higher voltages for future space applications. The converter design should be improved to reduce void space in the reactor core which would result in higher specific power systems. The cesium reservoir should be designed into the converter using a cesium graphite compound that is self regulating at coolant or collector temperatures. Also the effects of small amounts of oxygen in the cesium gap on converter performance should be determined by an applied research and technology program. Converter designs that combines the function of the metal-ceramic seal and the sheath insulator to reduce void volume in the reactor core and to increase converter lifetime at higher system specific power should be investigated. Also converter designs should be developed that improve the disposition of fission products, particularly venting fission gases out of the fuel.

One of the strongest recommendations from this working group is that there should be more interaction between the Thermoelectric (TE), Thermionic (TI), and Power Electronic (PE) technology efforts. The applied research and technology programs for all three technologies should be technically reviewed jointly at least once a year. The plans for each technology should be made jointly to ensure that each effort understands what the other technology efforts have accomplished and how it applies to their own efforts. The technology approaches that were suggested for combined programs are:

- 1) Combine thermionics and thermoelectric conversion into one conversion device
- 2) Use TE material as the emitter to collector lead in the TI converter to produce power rather than a loss of power

- 3) Use solid state electronic switching in a series connected string of thermionic or thermoelectric converters to produce high frequency AC power
- 4) Investigate the use of Si-C, B-C, La-S and Si-Ge materials to produce high power, high temperature and high radiation resistant power electronic components, and
- 5) Investigate the use of the thermionic electric insulators as a high temperature insulator in power electronic components.

CONCLUDING REMARKS

The TE, TI and PE technology working group recommends the following applied research technology efforts be pursued in the near future:

- 1) Thermoelectric materials technology be pursued for Figure of Merits of greater than $1.0 \times 10^{-3}/K$ at hot junction temperatures of 1100 to 1500 K and cold junction temperatures of 600 to 1000 K.
- 2) The technology required for thermoelectric converters designed for high specific power (300 watts/kg) and low production costs be developed.
- 3) Thermionic fuel-emitter and sheath-insulator technologies be pursued to develop methodologies for predicting emitter diametrical growth as a function of operating temperature, time and power density and for predicting electrical insulator degradation as a function of operating time, temperature and voltage gradient.
- 4) The technology required for high performance, small emitter diameter converter and minimum reactor core void be developed that is applicable for high power (1 to 100 MWe) electric power space systems.
- 5) The Power Electronics (PE) technology work closely with Thermoelectrics (TE) and Thermionics (TI) to develop high power, high temperature radiation resistant electronic switching components.

TABLE 1. SPACE POWER WORKSHOP GENERIC
QUESTIONS FOR ALL WORKING GROUP SESSIONS

1. What are the needed characteristics of your technology to enable the missions in the three sectors (military, public, commercial) in the period 1995 - 2005?
2. What missions are constrained by the existing level of the technology?
3. What are the technology constraints for the missions identified in question 2?
4. What mission(s) is (are) the strongest driver(s) for the technology?
5. What are the current significant technology deficiencies and critical barriers in your technology area?
6. Where are new technology approaches needed? Will these approaches provide enabling or enhancing technology?
7. What is the adequacy of current and planned programs?
8. What additional tasks should be done? What are the benefits? What is the necessary timing?
9. What are the lowest priority items in the present and planned programs?
10. What potential problems do you foresee regarding the interaction between your hardware and the environment that may be encountered in use? This can include contamination by mission operations as well as the natural environment.

TABLE 2. SPACE POWER WORKSHOP SPECIFIC
QUESTIONS THERMOELECTRICS AND THERMIONICS TECHNOLOGIES

Scope: Thermoelectric and thermionic devices and electric generating subsystem for use with a nuclear reactor. Thermionic may be in-core or out-of-core. Do not include reactor technology.

1. What is the threshold for success for reactor thermoelectrics, e.g., hot junction temperature and Figure of Merit? What thermoelectric materials might achieve these?
2. Is there a maximum reasonable power level to which thermoelectrics can be applied?
3. Is power processing a critical problem for large thermoelectric systems?
4. What are the critical technology problems for in-core thermionics and how can they be overcome?
5. Same as 4. for out-of-core thermionics.
6. What is the minimum emitter temperature for which thermionics give satisfactory performance?

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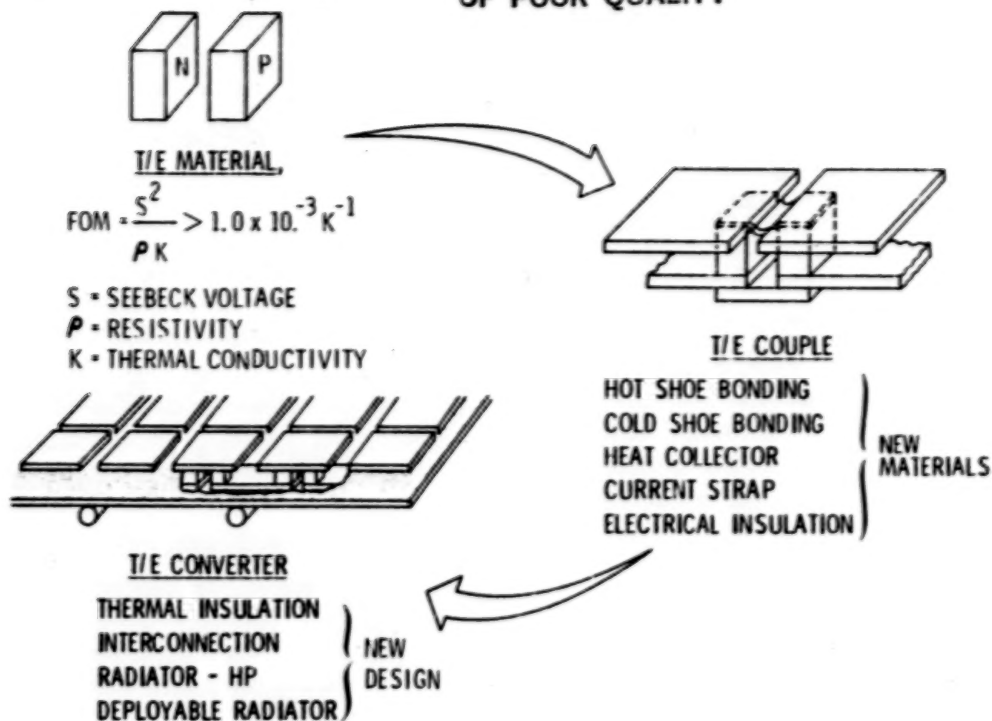


Figure 1. Thermoelectric (T/E) Technology Issues

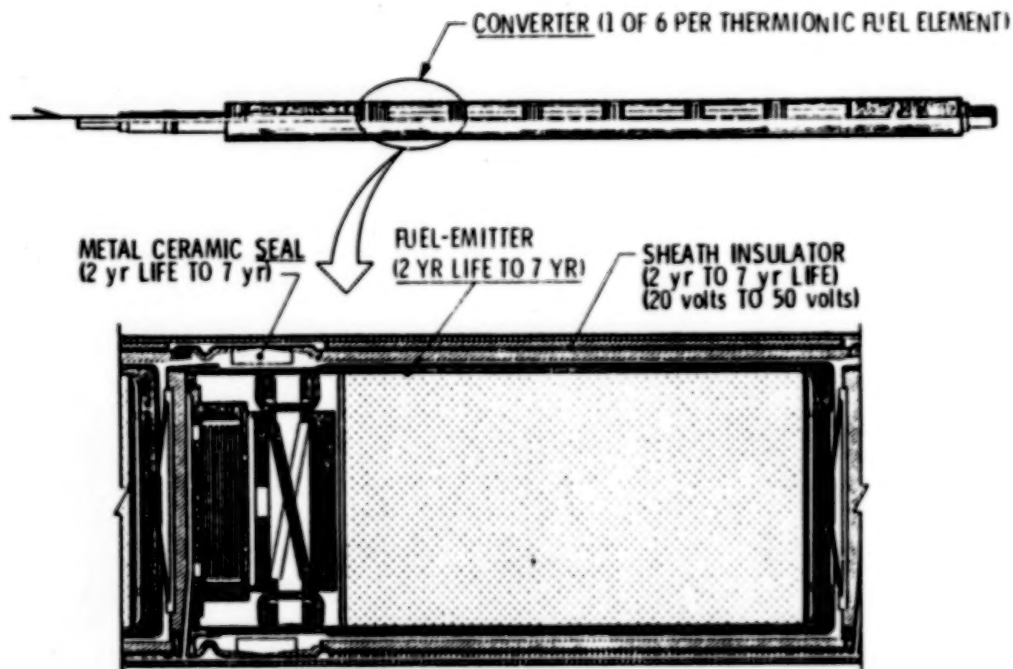


Figure 2. Thermionic (TI) Technology Issues

HEAT REJECTION WORKING GROUP SYNOPSIS

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There currently exists a coherent thermal technology plan which has been developed by the NASA Manned Space Station Steering Committee Thermal Working Group. This plan adequately addresses the requirements for technology development in the low temperature ranges (0° to 80°C) for incorporation into the Space Station. However, development goals in the moderate and high temperature ranges used by dynamic power systems and directions for thermal technology beyond 1995 have not been accurately charted. The Heat Rejection Working Group has attempted to better define the adequacies or inadequacies of the present technology program and provide a conceptual framework for future planning.

HEAT REJECTION - MISSION NEEDS

The distinct missions to be accomplished in space were perceived to be the following: Space Station, planetary exploration, commercial, "very high power", and military. Due to the relatively abbreviated time-span which was allocated for the Working Group to convene, the commercial mission area was not adequately addressed, but was judged to present no thermally unique problems which were not covered in the other mission schemes.

Requirements for heat rejection are tied closely to the power system which is utilized onboard the spacecraft. Preliminary Space Station planning has indicated photovoltaics will provide the primary energy with fuel cells or regenerable fuel cells acting to supplement required power during solar array occultation. This implies the use of low to moderate temperature radiators for rejection of waste heat. However, if growth versions of the station include solar dynamic power systems, radiator temperatures climb into the upper-moderate or lower high-temperature ranges. (Figure 1 illustrates the temperature range divisions which are being alluded to.)

Planetary exploration missions exhibit relatively low energy requirements (less than 1 kw) and therefore low total heat rejection loads. However, widely varying and extreme environments

will radically impact thermal efficiencies. This is contrasted by the "very high power" missions (such as radar imaging, lunar base, materials processing facilities) which will require large and dimensionally compact power sources. In such missions, moderate to high temperature heat transport and rejection will be required to disperse large quantities of waste heat.

Military missions will conceivably utilize the full range of the power scale but with the added capability of being "threat survivable".

KEY TECHNOLOGIES

The various mission needs infer the required thermal technologies which must be developed to meet these needs. Some of these technologies have been, or are currently being developed, however, some have not been approached due to the lack of previous driving requirements.

Space Station Missions

The key thermal technologies as they will be existing in near-term 1985 and as they are desired to be by 1995 for the Space Station are shown in figure 2. The heat rejection subsystem for an IOC (initial operational capability) Space Station will, at the minimum, have heat pipe elements which can reject approximately 2 kw of thermal energy at roughly 65°C (150°F) for a space constructable radiator. The demonstrated fin effectiveness of 0.6 causes radiator areas to be large, thereby distinguishing this as a factor which must be improved upon. Elevated radiator temperatures (100-140°C) which would accompany solar dynamic systems imply the development of moderate temperature radiators, not in the current thermal technology program plan.

Thermal transport and heat acquisition technology will still be predominantly Shuttle state-of-the-art in 1985, however, proof of concept testing will have demonstrated the feasibility of utilizing two-phase fluids to enhance the performance of these subsystems. It is envisioned that by 1995 two-phase thermal technology will have advanced to the point where it will be used throughout the Space Station, with the possibility of a nontoxic two-phase fluid being used internal to pressurized volumes.

Dynamic power systems would open up the capability of utilizing moderate to high temperature waste heat by experiments or material processing equipment aboard the station. The users of such waste energy have not yet been adequately defined, but the availability of this energy may stimulate greater interest in its utilization. But here again, as in the elevated temperature heat rejection, moderate and high temperature acquisition and transport technologies are not currently being investigated under the NASA thermal technology program plan.

Planetary Mission

Spacecraft which are targeted towards planetary destinations have typically utilized nuclear power sources. Though these spacecraft are relatively low in total power usage, adverse thermal environments have created unique heat rejection problems. Figure 3 indicates existing and desired thermal control technologies for planetary missions.

Thermal coatings regulate absorbed flux levels, but current coating technology is insufficient for long duration missions due to degradation in the solar absorptivity. An enhancement in thermal coating technology would benefit all spacecraft by stabilizing the optical properties and thermal predictability.

The various planetary missions of fly-by, probe, and lander present different thermal control design requirements, but are generically characterized as requiring high-performance insulations and heat sinks. Current insulations and sinks are not adequate for high temperature or high pressure environments, so development is required to enable full exploration capability of the inner and outer planets.

"Very High Power" Missions

Space missions which require very high power levels are growing as the concepts of what can be accomplished in the space environment are expanded. High power levels are accomplished through systems which typically utilize high temperature levels. Though large amounts of power can be generated by photovoltaics, more concentrated energy sources will mostly likely be utilized to reduce deployed areas, weight, and orbital maintenance requirements.

Though high temperature thermal systems have been investigated as early as the 1960's, requirements for their use have not been strong enough to continue development on a significant scale. The SP-100 project has initiated a level of effort in high temperature heat rejection and transport (in particular, heat pipes), but this effort must be augmented in order to verify the technology to enable other high power missions such as a growth Space Station, lunar base, and radar imaging satellites.

Heat pipe radiators and thermal transport concepts which operate at greater than 300°C (530°F) are the technologies required for

the high power systems to become more viable. Such advances in thermal technology will necessitate better comprehension of transient phenomena in high temperature systems, more efficient component contact interfaces, and more sophisticated analytical methods.

Military Missions

Power systems which are, or will be, used in military space missions cover the full range from relatively low magnitude photovoltaic systems to ultra-high power nuclear systems. The unique requirement made on the heat rejection components used in conjunction with these power systems for military spacecraft is that they be threat survivable. That is, they must continue to function, although in a degraded mode, after having been impacted by laser, neutral beam, nuclear, or projectile weapons. Radiation hardening is the only existing method of defending against the concentrated beam threats, with limited consideration given to explosive threat protection.

Design of heat rejection systems for military missions would be greatly enhanced by more accurate thermal modeling of laser and neutral beam impingement effects. These analyses could be confirmed through ground and flight tests resulting in the development of more effective counter-measures. Examples of proposed defensive methods include spectrally reflecting coatings, "switchable" coatings, high temperature materials, ablating surfaces, and IR decoys.

HEAT REJECTION-RESPONSES TO SPECIFIC QUESTIONS

The Heat Rejection Working Group was presented with a list of specific questions (shown in figure 4) which were to be addressed during its deliberations. Though the members of the group felt there were many additional questions to be confronted in the thermal area, discussion was limited to responses to the specific questions due to time constraints.

Critical thermal technologies which are envisioned in enabling future space missions include the following: a) stable coatings or surface treatments with low ratios; b) high temperature heat pipe materials (working fluids, containers, wicks, and coatings); c) reduction in radiator weight; and d) development of deployment mechanisms and interface connections.

Stable coatings for long duration missions have been pursued over a number of years with only moderate success. Contamination of spacecraft surfaces tends to override advances in optical coating technologies.

There are critical barriers perceived in heat rejection systems as they relate to power systems, but they are not judged to be insurmountable barriers at the present state of understanding. Developing long life, high temperature heat transport and rejection systems will most definitely require a large amount of resources, but previous development work on these systems has shown no "show stoppers" except for inherent material temperature limits and compatibilities at the high temperature ranges ($>1200^{\circ}\text{C}$).

Weight and volume reduction in heat rejection systems is another critical barrier which must be overcome in order to reduce the impact of this system on spacecraft configurations. Large deployed areas having significant mass impact greatly on control requirements and the architecture of the space vehicle. Additionally, radiator weight can be a significant design parameter in dynamic power systems.

Heat pipe radiators have no theoretical limit to their size in microgravity conditions. Size limitations are imposed by deployment or on-orbit assembly approaches, launch packaging constraints, and by ground testing capabilities.

Single phase fluid radiators are inherently sensitive to micrometeoroid damage due to the long tube lengths which are required in the radiator areas. A single puncture would disable a large radiator surface area due to fluid loss. This is the major reason heat pipe elements have been determined to be more applicable to spacecraft radiator systems - puncture of an element does not disable large portions of radiator areas. However, heat pipe spacing, liquid tube spacing and wall thicknesses are driven to a degree by micrometeoroid/debris projection. It would be prudent to reevaluate these projections and to update design algorithms for both heat pipe and liquid loop radiator designs.

High temperature heat pipe radiators have the inherent problem of freezing of the working fluid at lower temperatures. This is not a problem in itself, but recovery of the pipe to a functional state is. During transient conditions a liquid metal type of heat pipe could contain three phases (solid, liquid, and vapor). The solid phase could potentially block mass transfer within the pipe resulting in an inoperative condition. Methods of dealing with this type of problem have not been well defined. Since high temperature heat pipes are instrumental in the development of dynamic power systems, a more in-depth understanding of transient conditions and analytical methods for dealing with them at the element and system level must be generated.

CONCLUSIONS

The consensus of opinion within the Heat Rejection Working Group was that heat pipes and two-phase thermal systems would complement the needs of future power systems more readily than would single phase fluid systems. Current technology goals are aimed at developing lower temperature two-phase heat acquisition, transport, and rejection systems for space vehicles; however, moderate and high temperature development is not currently being investigated in a vigorous manner. The level of activity must increase in order to enable future high power mission scenarios.

Heat rejection is emerging as a key power system driver due to the fact that radiator and heat transport component weights can be significantly greater than the power system itself. Subsequently, power system trades must be based on appropriate heat rejection technology at the time the system will be flown. That is, 1995 power systems should not assume 1985 radiator weights. It can be concluded that weight and volume reduction through radiator refinements are lower risk and more cost effective than increasing the efficiency of dynamic power systems with increased temperatures.

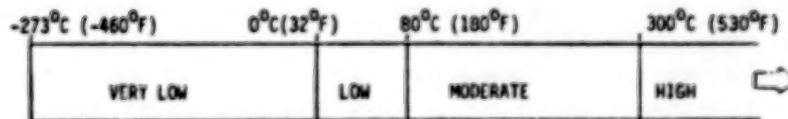


Figure 1. - Heat pipe/two-phase thermal transport temperature ranges.

KEY TECHNOLOGIES - SPACE STATION		
NEED	EXISTING 1985	DESIRED 1995
o LARGE RADIATORS	o 2 kW ELEMENTS (50"x1") o SPACE ASSEMBLY o FIN EFFECTIVENESS ~.6 o WEIGHT 5 Kg/m ²	o HIGH TEMP. FOR SOLAR DYNAMIC o ROTATING JOINTS o HIGHER PERFORMANCE HEAT PIPES AND FIN EFFECTIVENESS o WEIGHT ~2 Kg/m ² o SPACE CONSTRUCTION/DEPLOYMENT/RETRACTION
o HEAT TRANSPORT	o SINGLE PHASE (SHUTTLE) o TWO PHASE (PROOF OF CONCEPT)	o TWO PHASE - VARIOUS FLUIDS (WIDER TEMPERATURE RANGE) - VARIABLE TEMP. SET POINT - NON TOXIC FLUID - THERMAL STORAGE
o HEAT ACQUISITION	o SHUTTLE COLD PLATE (2w/cm ²)	o HIGH CAPACITY PLATES (2w/cm ²) AND HEAT EXCHANGERS o LOW ΔT CONTACT HEAT EXCHANGERS o REUSABLE FLUID CONNECTIONS o MAINTAINABLE EQUIPMENT INTERFACES

Figure 2.

KEY TECHNOLOGIES - PLANETARY		
	EXISTING - 1985	DESIRED - 1995
o FLY BY	o COATINGS EXHIBIT EARLY DEGRADATION	o DURABLE, LIGHT WT., ELECTRICALLY CONDUCTING INSULATION SYSTEMS o LIGHT WT. RADIATORS (VENUS PAPER)
o PROBE	o PARAFFIN HEAT SINKS o CURRENT INSULATIONS FOR VENUS DESCENT ARE MARGINAL	o HIGH PERFORMANCE HEAT SINKS o HIGH PRESSURE, HIGH TEMPERATURE INSULATION SYSTEMS (VENUS)
o LAUNDER	o PARAFFIN HEAT SINKS	o HIGH PERFORMANCE HEAT SINKS o HIGH PRESSURE, HIGH TEMPERATURE INSULATION SYSTEMS (VENUS) o REFRIGERATION SYS. FOR VENUS AND MERCURY

Figure 3.

HEAT REJECTION

Scope: Power system heat rejection subsystems and components. Include "conventional" fin tube and heat pipe radiators, but not advanced concepts like liquid droplet and belt radiators. Include materials, heat transfer, coatings, fabrication, stowage and deployment, meteoroid penetration.

1. What are critical technologies needed for fin and tube radiators and heat pipe radiators? Categorize by temperature level and power level.
2. What are critical barriers to overcome to achieve heat rejection system life of 7 - 10 years?
3. What are the size limitations for heat pipe radiators and how can they be overcome?
4. How well is the micrometeoroid penetration problem understood? What more knowledge is needed?
5. Are there critical technologies related to the heat rejection subsystem (heat pipe or fin-tube) following operational modes in a dynamic conversion system (startup, shutdown, cold soak, restart, part power, transients, etc.)?

Figure 4.

ELECTRICAL SYSTEM TECHNOLOGY WORKING GROUP (WG) REPORT

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NASA Goddard Space Flight Center

The charter of the Electrical System Technology WG was to assess the technology needs for space power systems (military, public, commercial) for the period 1995 to 2005 in the area of power management and distribution, components, circuits, subsystems, controls and autonomy, modeling and simulation.

The initial activity of the WG focused on two areas. First, the power levels needed for the 1995 - 2005 time period was discussed. These are summarized by sector in the first figure. There was general agreement that the military requirements for pulse power would be the dominant factor in the growth of power systems. However, the growth of conventional power to the 100 to 250kw range would be in the public sector, with low-earth orbit needs being the driver toward large 100kw systems.

The second WG activity led to the development of an overall philosophy for large power system development. These are summarized in the second figure. The approach to developing large kw systems that have lifetime of tens of years requires a radical departure from our 25 years of heritage in power system development for free-flying spacecraft. The approach must parallel the utility grid network that has evolved in the USA. In recognition of gross simplification, providing power to a multiplicity of users must be as simple as plugging in to the standard 120 volt 60Hz cycle outlets in your home. The question is not whether large systems are ac or dc, but rather at what point you provide ac to the user. The problem of common mode, grounding, EMI, EMC, etc. identified during real time integration of present systems must be solved "by design" of the multikilowatt system.

A very important feature of large kilowatt power systems is the ability to grow in an evolutionary manner. This is another way of saying that the basic system should be technology transparent, i.e., readily accommodate new system components as they evolve to flight maturity status. It is apparent that even the lower levels of multikilowatt power systems (10 to 75kw) will be hybrid in that the components such as batteries, fuel cells, and flywheels may all be used individually or in combination for energy storage. A similar argument can be made for planar solar arrays combined with concentrated arrays or solar dynamic components. A multiplicity of conflicting requirements will necessitate such a design approach if the power system is to be manageable and reliable. This, in fact, is the essence of the third guideline of figure 2.

The fourth guideline is based on the premise that there will continue to be a number of spacecraft required to meet the wide range of science and application requirements. The issue is one of modularity of the power system for serviceability, maintainability and evolutionary growth. Modularity of power systems is an expansion of the existing Multimission

Modular Spacecraft (MMS) where the power, data, and control subsystem are modularized for on-orbit replacement as recently witnessed by the repair of the Solar Maximum Satellite. Past program practices have been to "spare" at the component and/or assembly level. We are moving into an era where spares will be at the modular level.

In the interest of obtaining maximum output in a short period, the WG was divided into subworking groups to focus on specific technical issues. Each subgroup reported back to the WG on their findings. The following, with some editorial changes, is the report of each subgroup chairman.

Power Management and Distribution

A desirable characteristic of future high power military and NASA spacecraft would be common bus voltage characteristics to minimize component development costs. A further consideration is the impact of international cooperative ventures such as space station. Power bus characteristics must be defined to ensure compatibility of modules joined together in space for the first time. We recommend that an international committee of experts, including the user community, be tasked with developing an orderly time-phased set of guidelines, criteria, standards and specifications for future high voltage, high power buses.

While it may be impractical to test complete high power subsystems in an integrated manner, it will be necessary to test high power subsystem components up to megawatt class in order to verify subsystem computer model data bases. We recommend that planning begin for a national space power test facility, to be available to both the government and contractors, for up to megawatt level components testing.

We are convinced that improved thermal control is vital to achieving longer lived high power components. This will involve much deeper penetration into power equipment configurations by thermal designers. Integral heat pipes and fluid loops within power subsystem components may be required to maintain acceptable temperature on future space power systems.

Further improvement in availability will be accomplished through system level automated management techniques which optimize system utility in the presence of faults.

Components, Circuits and Subsystems

We feel that technology is available for multihundred kilowatt class power systems. However, considerable uncertainty surrounds technology readiness for megawatt class systems. We have concern that commercial high power components may not be available in high reliability versions for space station class missions. We recommend effort to assess high power component piece part availability status and the process of upgrading these parts to spaceflight quality begin in the near future.

The most critical concern of the component technology specialists was lead time. Components required to build electric power systems of the 1995 to

2005 vintage must be developed, evaluated, and proven to the satisfaction of system designers before the mid 1990's. We are, therefore, faced with the problem of developing components for system hardware which is not yet defined or left with the alternative of developing advanced systems based on whatever components are available. Early recognition of this basic truth permits us the opportunity to define and resolve the critical issues shown in figure 1.

Component life and availability of components are concerns for both component and system development. Component life is affected by the stresses, environmental and electrical, which it must endure. Low cost and ready availability require high production volume. Components developed must, therefore, be attractive to the commercial market. Components which are unique to the space application will require development of specialized contractor capability and suffer both cost and logistics penalties.

Components other than semiconductors which must be developed for the system to be defined under the critical issues include:

- high voltage fuses,
- rotating joints,
- high voltage connectors and terminators,
- high voltage, high frequency power distributor transmission lines, and
- remote power controllers for HV and HF

Controls and Autonomy

Power system autonomy is enabling for future space systems over the entire power range from kilowatts to tens of megawatts. There are alternative sources of energy and alternate energy storage concepts, but there is no alternative to incorporating autonomy. It is the responsibility of the power discipline. Furthermore, autonomy will contribute to technology transparency, increase reliability, and reduce cost.

The key outputs of an autonomy program will be a detailed analytic understanding of the elements of power systems, design guidelines, and architectural constraints; but the most important product will be a cadre of power system autonomy expertise that will be the repository of technical knowhow. One of the key items that was identified by the working group is the need for a detailed review of work that has been done in advanced aircraft designs, in nuclear submarines, and in the utility industry. In addition to assuring that we know the lessons learned in other power autonomy applications this might initiate the building of the recommended cadre.

We need to define programs that evolve the benefits from state-of-health monitoring including preflight checkout, insipient failure detection, reconfiguration, and post flight analysis. We need to understand the impact on operations and mission planning, particularly as related to maintenance and module change-out schedules. The path from the present SOA to this future ability is via an incremental buildup from the control of individual components and the subsequent synthesis of these products into

the larger more complex systems. Detailed studies of this kind will surface relationships between component parametric characteristics and component health. There will also be a need for optimization analysis so that the maximum system reliability can be projected. The potential impact on technology transparency and on the relief in component quality control level needs to be reviewed.

This technology area needs a long term consistent level of support. It needs to be worked with industry on a continuous basis. Clear cut near-term, mid-term, and far-term goals need to be established. Because this area inherently deals with less tangible and items (algorithms, architectures, and design guidelines), the goals need to be carefully chosen and monitored. The area needs a strong technical leader.

Modeling and Simulation

It has become clear that a generic technology need for space power systems of all sizes involves modeling and simulation. With such a tool in hand, designers, analysts, and operators of power systems can understand the tradeoffs, interactions, and predictions concerning power. To do this, the following capabilities will be required: scale models that predict performance, model simplification that appropriately depicts the level of detail required for a specific problem (possibly using expert/smart systems, etc.), and model verification. The use of downsized scale models will facilitate checkout of designs and control laws. Such a model (although smaller) should allow the designer the ability to verify design performance, and analyze various failure mechanisms in a way that will not jeopardize flight hardware.

At the same time, analytical models will continue under development. While component modeling is well underway, very little is understood on what level of detail is needed to solve various levels of problems. For energy balance issues, it should not be necessary to model the switching of the various power conditioners. However, such switching issues are critical for dynamic and transient studies. The use of expert systems in such models will allow the appropriate level of complexity for the model to be determined by the tool and be completely transparent to the user.

As in all modeling and simulation tasks, verification of the model and results are critical. This will need to be done for the detailed subsystem models and for the simplified system models. Confidence in the models will insure that the concerns for full-scale testing can be reduced. As long as computer or scale models can be used to predict performance, only model verification tests need be performed on the full-up system.

A key development area for simulation is computer hardware, software, and firmware. These fast-paced technologies will need to be monitored and advances incorporated as appropriate into the power system model.

Finally, it was the group consensus that the current modeling work be continued, built upon, and expanded so that the above capabilities are incorporated. To insure this, a top level "Executive Program" needs to be identified and specified.

SECTOR	POWER LEVELS		
	1985-1990 (kW)	1990-1995 (kW)	POST 1995 (kW)
MILITARY (CONT.)	2-12	12-30	30-100
MILITARY (PULSE)	-	50-250	1000
PUBLIC (NASA & LEQ)	2-15	2-150	100-250
COMMERCIAL (GEO)	2-4	2-8	10 TO 20*
INTERPLANETARY	-	-	25-100 kW

Figure 1.

GUIDELINES FOR DEVELOPMENT OF LARGE SPACE POWER SYSTEMS

- o PROVIDE MAXIMUM BENEFIT TO MAXIMUM NUMBER OF USERS
- o HAVE EVOLUTIONARY GROWTH POTENTIAL
- o HANDLE SEVERAL POWER SOURCES AND ENERGY STORAGE
- o HAVE COMMON MODULES FOR SEVERAL DIFFERENT SPACECRAFT TYPES

Figure 2.

POWER MANAGEMENT AND DISTRIBUTION

- o STANDARDS INTERNATIONAL COMMITTEE, INCLUDING USERS, TO DEVELOP TIME-PHASED GUIDELINES, CRITERIA, STANDARDS AND SPECIFICATIONS FOR POWER BUS, (WAVEFORM, FREQUENCY).
- o VERIFICATION TEST AND SIMULATION
 - NATIONAL FACILITY TO TEST PM LEVEL SUBSYSTEM ELEMENTS
 - BUS SIMULATION MODELS FOR SUBSYSTEM DESIGN VERIFICATION
- o LONG LIFE/
HIGH
AVAILABILITY
 - INTEGRAL ELECTRICAL/THERMAL DESIGNS DOWN TO PIECE-PART LEVEL
 - SYSTEM AUTOMATED RECONFIGURATION TO MAXIMIZE UTILITY AFTER FAULTS
- o BARRIERS
 - MATERIALS/ENVIRONMENT DEVELOPMENT USING SPACE STATION AS A LAB
 - SPACE QUALIFICATION OF AVAILABLE/EVOLVING DEVICES

Figure 3.

COMPONENT KEY TECHNOLOGY ISSUES

- o SYSTEM ARCHITECTURE
- o HIGH VOLTAGE, FREQ. PROBLEMS
ARCING-CORONA - PROTECTION - ISOLATION
- o ENVIRONMENT - INSIDE - OUTSIDE
- o PACKAGE/COOLING
- o FLIGHT QUALIFICATION

LONG LIFE - AVAILABILITY

- DEFINE AND CONTROL THERMAL/ELEC. STRESS
- COMMERCIAL COMPONENTS - MULTIPLE USES
- DEV CONTRACTOR CAPABILITY FOR UNIQUE DEVICES

CRITICAL COMPONENTS

- HV FUSES/RPC
- CONNECTORS
- TRANSMISSION LINES
- SWITCHING DEVICES

MODELING

- BETTER COMPONENT MODELS REQUIRED

Figure 4.

AUTONOMY AND OPERATIONS

- o IMPORTANT AREA FOR FUTURE SPACECRAFT
- o AUTONOMY - COMPLETE INDEPENDENCE
- o CADRE OF EXPERTISE - DESIGN INCREMENTAL - SOLUTION AND INCORPORATION
- o SENSORS AND ALGORITHMS
- o ASSOCIATE MODELING AND UNDERSTANDING

Figure 5.

MODELING AND SIMULATION

KEY TECHNOLOGY ISSUES

- o DEVELOPMENT OF A COMMON EXECUTIVE PROGRAM FOR SIMULATION (ALL LEVELS)
- o DEVELOPMENT OF SCALE MODELS FOR LOADS, SOURCES, DISTRIBUTION SYSTEM
- o VERIFICATION (TEST FACILITIES, DATA BASE)
- o DEVELOPMENT OF MODEL SIMPLIFICATION TECHNIQUES USING EXPERT SYSTEMS, ETC.
- o UNDERSTANDING OF DIGITAL/ANALOG INTERACTION IN "REAL" SYSTEM VERSUS SIMULATION

Figure 6.

ENVIRONMENTAL INTERACTIONS WORKING GROUP REPORT

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Environmental interactions is a term that is applied to interactions between spacecraft systems and the space charged-particle environments at all altitudes from low-earth orbits to interplanetary space. The detrimental effects resulting from such interactions range from systems upsets to failures, from material surface degradation to material loss and from power drains to transient shutdowns. Technology investigations have been conducted since the early seventies to study these interactions. These analytical and experimental investigations, supplemented by shuttle experiments, provide the data base for understanding environmental interactions.

Interactions of Concern

The interactions of concern identified by this working group are given in Table 1. A brief discussion of each interaction is given in the following paragraphs.

Transient Environment Models

After many years of measurements of the space environment parameters, there are reliable models of an average, static environment. However, these models do not treat temporal or spatial variations in detail. The data base is inadequate, at the present time, to support the development of this transient environment specification. Yet, temporal environmental effects can be important in understanding the interactions between the space environment and the large space systems proposed for future missions.

Large Space Structures

As structures become large, they can influence the surrounding environment as well as being influenced by the environment. A large body moving through the space plasma environment creates a strong wake effect which can result in noise generation. This was noticed in shuttle experiments. The noise could impact operations of communications systems or others sensitive to electrostatic and electromagnetic noise.

The proposed large structures can be both a source of outgassing material as well as a possible sink for outgassing contaminants. Contamination is known to be a serious concern on small satellites and its effect increases with size. The larger bodies may require more stringent altitude control maneuvers resulting in additional particulate contamination.

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Solar Array Space Power Systems

The proposed space missions generally require large power generating capabilities. The space station baseline has a solar array capable of generating 200 kw so that the 70 kw bus power can be provided. For these large power levels, the bus voltage is being increased beyond the present 30 to 60 volt values. However, as the voltage increases, then environmental interactions such as breakdowns and power drains become a serious concern. The large, high voltage solar array power system will control the potential (relative to space) of any large structure. This implies possible differential charging between a station and a docking body, which could lead to a safety problem. The electric fields surrounding the body could enhance contamination.

The environmental interactions with a high voltage solar array are serious. They range from transient shutdowns of the array due to breakdowns to manned safety questions. What is a "safe" operating voltage for such an array? This question can be answered only based upon preliminary data; possible interactions have still not been identified and understood.

Nuclear Space Power Systems

Environmental interactions with nuclear power systems have not been studied in any detail. However, the descriptions of such systems usually quote multi-kilowatt power levels which implies a high voltage distribution system. This would indicate high voltage problems similar to those discussed above. The nuclear source radiation could provide the most severe environmental conditions.

Single Event Upsets

Digital logic components have a demonstrated ability to switch state as a result of ions passing through the semiconductor material. As spacecraft become large, more logic circuits are used, increasing the probability of such upsets. These upsets can cause effects ranging from nuisances to major disasters and are difficult to prevent. Each component must be evaluated to determine if an upset would cause a serious system malfunction. If it would, then corrective techniques must be employed.

Material Degradation

The ram-oxygen degradation of kapton has pointed out that there can be unsuspected material erosion processes that would exist in space. While engineering solutions may be forthcoming for this problem, these techniques may not be adequate for other material interactions. In addition to chemical processes, there is the damage due to sputtering which would be enhanced in large structures and by biased surfaces. The material that is eroded will return to the spacecraft increasing surface contamination.

Other Planet Missions

The missions to other planets discussed in this workshop seem to be characterized by multi-kilowatt power requirements and will use ion drive for propulsion. The high power levels will introduce the same problems that have been previously discussed (i.e., high voltage power generation). The ion drive will increase the local plasma environment - the mission will carry its own environment along with it. In some ways this is beneficial while it may be detrimental in others. The total system concept must be evaluated.

Planetary atmospheres could introduce environmental interactions for probes or lander missions. Lightning due to triboelectric effects could be a real concern during descent. Single event upsets can also be increased during the mission.

A lunar habitat mission should be concerned with a contamination problem that could arise from charged lunar dust.

Summary

A brief review of possible environmental interactions has been conducted as part of the Space Power Workshop. It has been found that a technology program to investigate these phenomena has been underway since 1980 to support what was then conceived as the 1990's missions. Mission planners have moved much more rapidly than the technology growth in that they now require information and capabilities that will not be available till late 1980's.

The environmental interactions should be recognized as serious concerns for the designs of large space systems. Some of the more prominent interactions have been identified here but this list is not intended to be all-inclusive; there is still much that is both unknown and unsuspected in interactions with the environment. However, even though these interactions are complex, they are logical and can be understood and kept from being detrimental.

Recommendations

The following recommendations are presented for both near-term and far-term considerations:

Near-Term

1. Upgrade the existing technology plan to make its schedule more compatible with mission plans. Increase funding to accelerate progress.
2. Start process of working with designers/engineers on specific programs. Continue generic technology to develop the tools and techniques needed.
3. Incorporate environmental sensors on all shuttle flights to obtain a better data base for temporal environmental model.

Far-Term

It is difficult to project far-term requirements when the near-term needs have not been fulfilled and the technology is still developing. However, the following appear feasible:

1. Investigate different power system concepts, such as generating at modest D.C. voltage and transmitting at A.C. levels or developing A.C. power generating concepts.
2. Future requirements may grow to the levels where pulsed power or very high voltage power generation and transmission is required. Interactions in these systems are not understood and should be studied.

Table 1

Interactions of Concern

- o Transient Environment Models
- o Large Space Structures
- o Solar Array Space Power Systems
- o Nuclear Space Power Systems
- o Single Event Upsets
- o Material Degradation
- o Other Planet Missions

ADVANCED AND NONTRADITIONAL CONCEPTS WORKING GROUP REPORT

A. Hertzberg
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The purpose of this report is to examine advanced and nontraditional concepts relating to future space power requirements with special emphasis on the requirements for the space station. The group was motivated by the estimated power requirements for a fully operational space station (100-200 kW), a manifold increase over the power requirements which can be conveniently met by current technology. The dominating effect of the power system on the design of the space shuttle both from a mass and volume indicated that new approaches which offered more than incremental changes may prove enabling. Therefore, the group concentrated on high pay-off areas related to heat rejection, energy conversion, and energy storage in particular. In the following sections, we will report on some of the key findings of this working group.

INTRODUCTION

The compelling need for growth in space power capability was a key note of this meeting. This is particularly important for the manned orbiting space station. While in principle, available technology can be utilized to meet these requirements, it is nonetheless clear that the space station slowly becomes dominated by its companion energy systems as power requirements grow to the 100 kW level. Indeed, the very viability of the space station can be questioned by the high cost of such energy systems which are required for several potential economic functions of the space station such as materials and pharmaceutical production. These comments in no way denigrate the excellent work that is now being carried out to improve the capability of our present power systems, which constitute a hard one body knowledge which is the only basis on which a systems manager can conceivably approach the problem of space station design. At the present time, however, the various questions raised in these design approaches suggest that it is important to examine the innovative and even risky concepts. In particular those which offer the promise of bringing the cost of power to a level that can be supported by a space station economy.

With this in mind, the advanced and nontraditional concepts working group reviewed some of the structure of ideas which are currently being examined and were discussed at the meeting. Due to the heterogeneous nature of the group and the limited time available, our discussions were truncated and are certainly not complete. In addition, the group concluded that by no means was the pool of high leverage new ideas exhausted and that advanced technology exploration must be encouraged to insure that we indeed, do have a future for space power systems. In particular, concepts that might aid a number of missions, in the opinion of the working group, should receive the most support.

It is unfortunate that there was not sufficient time available to the working group to carry out an indepth technical appraisal of these concepts or to make value judgments with a precision that they would have preferred. Also, there were several ideas suggested, such as direct launch concepts which appear to be particularly energy efficient in carrying bulk payloads into lower earth orbit or geosynchronous orbit which the severe time limitations only allowed introducing discussions.

In the following sections, as chairman, I will attempt to list some of the advanced concepts. This grouping, of course, must be heterogeneous, but bound by the common thread of concepts which do contain the potential of offering significant advances in space power technology.

CONCEPTS

- o Dynamic Radiator Concepts
- o Advanced Nuclear Reactors
- o Advanced Energy Conversion Cycles
- o Beam Power Transmission
- o Electrodynamic Tethers
- *o Direct Contact Heat Exchangers
- *o Thermal Storage
- *o Supporting Component Technologies
- o Advanced Technology Exploration (In particular basic research such as superlattice and surface plasmons, thermal photovoltaics, and photoelectric chemistry)

- * Due to the limited time that was available, these concepts were identified and were not presented.

DYNAMIC RADIATOR CONCEPTS (HEAT REJECTION SYSTEMS)

The critical nature of the thermal management of large space power systems becomes almost self-evident as we apply these systems to a complex habitable structure in space such as the space station. In view of the problem, several groups are currently examining dynamic radiator concepts. The preliminary indications of this research program have indicated that major benefits perhaps even order of magnitude reductions in mass and volume requirements may be achieved with such systems.

The working group as a whole felt that these must receive attention and encouragement as an aspect of enabling technology for future space systems. Below are listed two classes of these radiators, the liquid droplet stream radiator, and the moving belt radiator, with specific comments from the committee about their potential and key technology issues.

DYNAMIC RADIATOR CONCEPTS

DROPLET STREAM RADIATORS:

Heat rejection from a sheet of very small liquid droplets that are collected, reheated and projected into the stream, using low vapor pressure fluids

BENEFITS:

Low specific mass

Compact packaging for launch

Automatic deployment in space

No surface coatings

APPLICATIONS:

Moderate or large radiator requirements

TECHNOLOGY ISSUES:

Fluid selection, stream generation, lossless collection, mission restrictions, trajectory, control, environmental interactions, system study to define technology requirements, identification of flight experiment

MOVING BELT RADIATORS:

An endless belt loop with local heating and radiation from moving surface. Heating may be enhanced by low vapor pressure grease (contact surface)

BENEFITS:

Low specific mass

Compact packaging and simple deployment

Wide range of potential temperature levels and heat load

APPLICATIONS:

Any large thermal radiator requirement

Waste heat from heat engines

Thermal control of spacecraft

TECHNOLOGY ISSUES:

Performance data

Lifetime demonstration

Heating method, drum or direct contact

ADVANCED ENERGY CONVERSION CYCLES

The potential of advanced energy conversion cycles resulted in significant amount of discussion by this group due to the renewed interest in the Brayton cycle and in the Stirling cycle as potential energy conversion units of either nuclear or solar powered thermal energy systems. These, in particular, offered apparent benefits when combined with advanced nuclear power generating systems and as such the working group feels that they invite further study. Some of these are listed below along with their potential benefits and key issues. As stated previously, there was little time to examine these in the detail desired. The single loop, (Helium) Brayton cycle (non-regenerated Brayton) for use with very high temperatures, 1500 K+ gas cooled reactors, appeared to be an opportunity to exploit the high temperature potential of nuclear energy systems.

SINGLE LOOP (HELIUM) CLOSED CYCLE BRAYTON (NON-REGENERATED)

USING HIGH TEMPERATURE (1500K+) GAS COOLED REACTOR

BENEFITS:

Compact, low mass design because of high power density eliminates heat source (liquid to gas) heat exchanger and recuperator

Reduces cycle pressure losses

Improves overall system reliability through the elimination of 'failure prone' heat exchangers

Eliminates the need for liquid loop pumps

Reduces radiator area required

TECHNOLOGY ISSUES:

High temperature gas cooled reactor development needs to be resumed (use of graphite material)

Tradeoff studies are needed on potential increase of volume of gas cooled reactors as compared to liquid metal reactors

ADDITIONAL ADVANCED ENERGY CONVERSION CYCLES

EXAMPLES:	Ericsson cycle
	Reacting gas cycles
	MHD power conversion
	Closed cycle gas reactor
BENEFITS:	High specific - improved efficiency
	Reduced heat rejection radiator requirements
	Nuclear or solar heat source
TECHNOLOGY ISSUES:	Ericsson - no barriers
	Reacting gas-materials, working fluid selection-system design
	MHD generator design, materials
	High temperature gas reactor design

ELECTROMAGNETIC BEAM TRANSMISSION

This meeting uncovered specific applications for beam power transmission which may enhance the potential of these systems when applied to the space station. While originally conceived for the purpose of beaming solar energy to earth or other spacecraft, there now appear to be intermediate applications which might use these systems to advantage. For example, as we increase our power demands in geosynchronous orbit, the battery mass necessary to insure continuous coverage becomes a dominating part of the space power systems requirements, despite the fact, that the eclipse period rarely lasts more than one hour and occurs at the most only a few times a year, it would then seem that a beamed energy from a laser (even earth based) may prove cost effective in filling in for the sun during these brief periods. The capital investment structure need not be large since, due to the short duration of the power requirements, system efficiency requirements can be met by existing technology. This would appear to present a unique opportunity to enhance the application potential for beamed laser energy capability of exciting a photovoltaic systems. In addition, relatively short wave length, microwave transmission, may well prove the most effective way of beaming energy from a companion satellite nuclear power plant to a manned orbiting station. Many of the critical problems dealing with safety shielding would be circumvented by this nontethered approach.

CONCEPTS

LASERS:

Solar, nuclear, thermal and electric powered; long range capability with small optics transmission and basic research benefit from DoD programs

BENEFITS:

Separation of power source from power consumer

Mechanical simplification of high-power spacecraft

Continuous power in orbit (minimum on-board storage)

APPLICATIONS:

Powering spacecraft in high-drag and high-radiation orbits

Continuous power to lunar base and lunar rovers

R² shielding from nuclear reactor

Power for electric and thermal propulsion

TECHNOLOGY ISSUES:

Shorter wavelength operation

High-intensity conversion

System efficiency

MICROWAVES:

Transmission distance \leq 100km

BENEFITS:

Permits separation from nuclear reactor

Mechanical freedom for spacecraft/power-system configuration

Eliminates power storage from power user

APPLICATIONS:

Nuclear reactor to user

Power to multiple users

TECHNOLOGY ISSUES:

Antenna and rectenna diameters

ELECTRODYNAMIC TETHERS

An interesting concept discussed by the working group was the concept of eletrodynamic tethers used as an energy storage device. Due to the significant mass penalties that appear to be created by battery or fuel cell storage concepts, this appears to be a bold but innovative approach that should be examined. It should be specifically pointed out that tethered concepts appear to be receiving considerable support from segments of the space community.

CONCEPTS

Electrodynamic tether provides a motor/generator function using the tether as an armature moving through the geomagnetic field to reversibly convert orbital energy to/from electrical energy.

Use 1/4 KV induced voltage in 5-20 km long gravity gradient stabilized wire in LEO to generate electric power for short term missions, or in reversible motor/generator operation on long duration missions to store electric power for day/night cycle or sustained peak load demands.

BENEFITS:

More cost effective power source for short-term missions, with lower consumable mass vs fuel cells, etc.

Orbit energy to electrical power conversion efficiency potentially 90-95%

Lower mass and more cost effective power storage (vs batteries) for long-term solar power missions requiring day/night cycle operation. Full cycle "charge/discharge" efficiency potentially 80-90%

Greatly increased peak power and "total depth of charge/discharge" capacity for LEO operations

Capacity for use with (Resistojet) electric rocket system to produce net power from disposal of waste water and Isp values of 400-120 sec.

Capability for orbit maintenance of large solar arrays in very low orbits using 1-10% of array power for ΔV thrust, eliminating present orbit maintenance fuel requirements

KEY ISSUES:

Performance of hollow cathode "plasma brushes" for current coupling between tether and ionosphere

Verification of expected variation of useful induced voltage with current flow through complete tether/ionosphere circuit

Refinement and experimental verification of tether dynamics under variable thrust/drag electrodynamic loads

Development of space compatible insulation and power processing electronics for operation at 1-4 kV

FLIGHT EXPERIMENTS:

Verify basic performance parameters, using low power (20 watt) in 200 meter centrifugally stabilized wire and LEO operation of hollow cathodes

Higher power (1 kW) verification of actual thrust/day deflection of tether

Verification of computer simulation of on-orbit dynamics

Scientific study of detailed ionospheric interaction mechanics

EXPLORATORY ADVANCED CONCEPTS

The working group briefly explored several concepts which are either in a basic or exploratory research phase. For example, there are indications of new direct launch concepts which may have important applications. In particular, with respect to the economy of a space station, in refueling and replenishment of consumables such as fuel, life support requirements and raw materials could be launched into orbit and recovered by a shuttle base tug to improve the economics of space station operation. In addition, there appear to exist several elegant advanced solar voltaic concepts which are still in the basic research phase and which appear to offer truly significant advantages.

ADVANCED SOLAR PHOTOVOLTAIC CONCEPTS

SURFACE PLASMON DEVICES:

Solar energy conversion by separating energy bands within the solar spectrum - then exciting surface plasmon waves which are processed and converted to bulk plasmas which transport energy to arrays of diodes which collect/extract the energy

BENEFITS: High efficiency-reduced solar capture area

APPLICATIONS: Space solar photovoltaic power systems

SUPERLATTIC PV DEVICES: Solar energy conversion concept using superlattices formed by alternate thin layers of III-V semiconductor materials with unusual electronic properties which can be varied independently to produce new PV concepts

BENEFITS: High efficiency
Low solar capture area
High radiation tolerance

APPLICATIONS: Space solar photovoltaic power systems

TECHNOLOGY AREAS-ISSUES: Both concepts are in basic research stage at this time

SUMMARY

The working group regards this report only as an example of the enabling ideas discussed in the time allotted, and we feel that the potential indicated is only a sample of the possibilities that should be explored. Further, the working group feels we should avoid the trap of continuing only to advance our incremental technology base at the expense of exploring the potential for new ideas. In the opinion of the group, both must be pursued with equal vigor. Indeed, despite the relatively undeveloped nature of most of these concepts, they appear to contain some aspects of merit or validity. The time limit proved frustrating and the working group and the chairman, feel that a special permanent working group should be set up to examine the structure of new enabling technologies and pursue these with vigor in the near future.

END

DATE

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